AIAA Foundation Student Design Competition 2022/23
Undergraduate Team – Engine

A Hybrid-Electric Propulsion System Using Fuselage Boundary Layer Ingestion for a Single-Aisle Commercial Aircraft

- Request for Proposal -

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Abstract

Many types of hybrid-electric propulsion systems are currently being investigated because of their potential application to sustainable aviation and their contribution to much-needed atmospheric benefits. Such engines must be integrated closely with the airframe. A leading contender from 2016 is the *NASA STARC-ABL* aircraft, which is powered partly by a single electric fan, located around the rear of the fuselage. The electric fan is driven by power extracted equally from two primary turbofans, mounted conventionally beneath each wing. These engines also provide the remainder of the thrust.

This Request For Proposal asks you to design a new hybrid-electric propulsion system for the *NASA STARC-ABL* with the same configuration. A significant feature of the aft fan is the ingestion of low-speed boundary layer air from the aircraft fuselage, so you are asked to discuss the merits and practical challenges that this concept presents.

The baseline engine for this study is a generic model of the *CFM56-7B24*, constructed from publicly available information. Details of this model – built at sea-level static operating conditions - are provided to assist you. Generation of your own version of the baseline engine is mandatory and is deliberately set to provide training and experience in generating a model that functions and looks right. Your baseline model will also be needed to obtain the thrust required for the new hybrid-electric system which is to be designed for cruise conditions at 35,000 ft, Mach 0.8.

Examine a select matrix of new hybrid-electric propulsion systems to determine the mass and performance trends in order to select your best candidate. Compare the performance and total fuel consumption of each of your new candidate hybrid-electric propulsion systems over a typical mission with that of the baseline engine model at the aircraft condition. Choose your best candidate, based on fuel burn over an assumed simple mission, while also considering the complexity and cost of your design. Finally, run your selected hybrid-electric engine off-design at sea-level takeoff conditions and compare the overall net thrust to the aircraft with that from two baseline engines.

This competition is intended to expose students to the trade studies and conceptual evaluations that are the foundation of gas turbine engine preliminary design. Showing evidence of a thorough design space study and justification for the final selected design will be more highly weighted than detailed assessment of a specific component.

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1. Introduction

1.1 The Aircraft

Interest and investment in hybrid-electric propulsion systems has grown substantially in the past ten years or so owing to their potential application to sustainable aviation and significant benefit to atmospheric conditions through fuel-savings. A leading contender from 2016 is the NASA STARC-ABL with a single large aft fan, located around the rear of the fuselage, which captures a large annular portion of the rear fuselage boundary layer [*1*]. The aircraft is shown in *Figure 1.1*, as well as on the front cover, and its propulsion system is the topic of this RFP.

Figure 1.1: The NASA STARC-ABL Aircraft

The aircraft is a single-aisle, 180-passenger commercial transport, with an entry-into-service date around 2035. It is a future version of a current *Boeing 737-800* or *Airbus A320*, powered by either two *CFM56-7B24*, two *IAE V2500* or two *Pratt & Whitney PW1000G* turbofan engines.

1.2 The Engines

We choose a generic model of the *CFM56-7B24* as our baseline engine. The model was constructed using *GasTurb 14*, based on data available to the public [*2*]. It is not especially accurate; several educated guesses and many trial and error iterations were used in its generation and the cold nozzle is the best that could be achieved currently with the software.

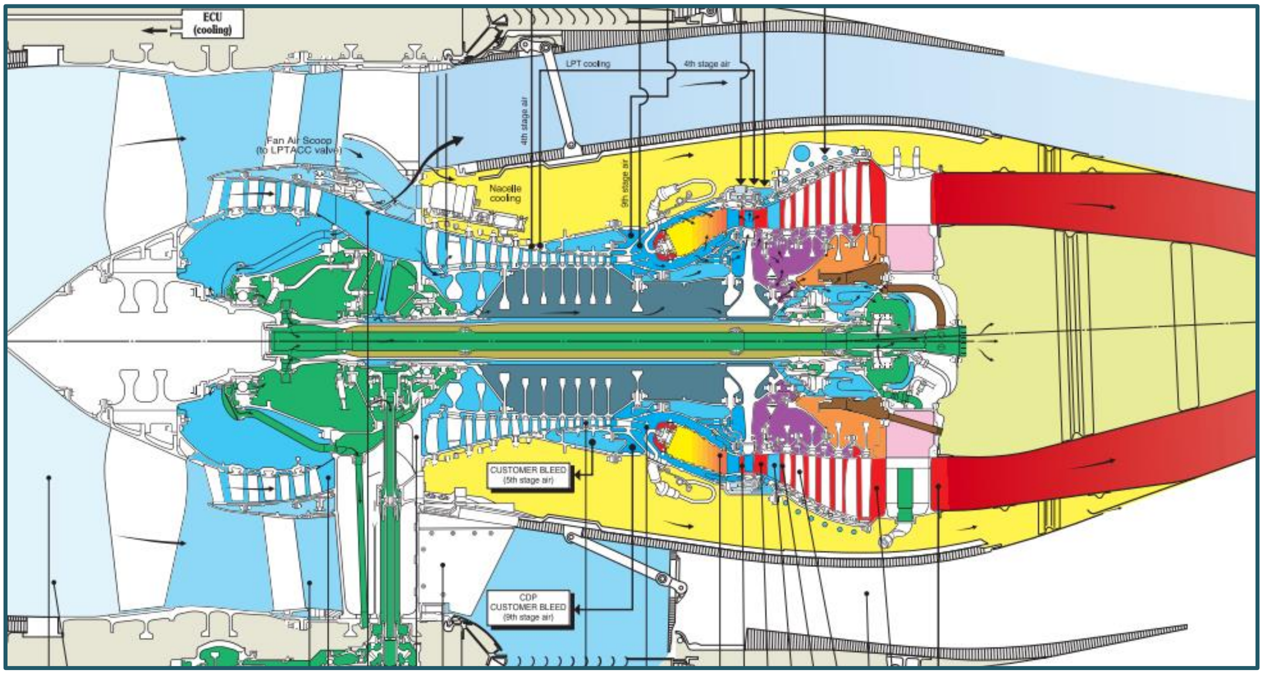

Figure 1.2: CFM56-7B24 Cross-Section

Figure 1.2 is a cross-section of the *CFM56-7B24,* which illustrates the flow path geometry, major turbomachinery assemblies, stage counts and the general levels of flow temperatures encountered. *Table 1.1* summarizes some major design features.

The overall length - 98 inches as published – is a “flange-to-flange” measurement. We know the fan tip diameter is 62 inches and we can estimate that the quoted length of 98 inches corresponds to the distance between A and B, the locations of the flanges indicated in *Figure 1.3,* upstream of the fan leading edge and downstream of the LP turbine rear frame. This is considerably less than what anyone would refer to as the overall length of the engine! So, as you can see, engine length can be interpreted fairly loosely! Even though we are always concerned with the accuracy of the models we produce, let’s not worry too much about that; we all know what we are trying to simulate!

The dry weight of 5432 lbm, published in *[2],* excludes the inlet, the tailpipe and the nozzle, so we will allow for this later in the discussion of *Table 3.23, Sub-section 3.10,* when we estimate the net mass factor that accounts for the secondary systems outside the flow path that are not accounted for directly in our preliminary design activity.

Engine Type	Turbofan
Number of Compressor Stages (Fan, Booster, HP)	1, 3, 7
Number of Turbine Stages (HP, LP)	1, 4
Combustor Type	Axial annular
Max. Power at Sea Level	24,000 lbf
Specific Fuel Consumption at Max. Power	0.37 lbm/hr/lbf
Overall Pressure Ratio at Max. Power	26
Bypass Ratio at Max. Power	5.3
Max. Envelope Diameter	65 in
Max. Envelope Length	98 in
Dry Weight Less Tailpipe	5,234 lbm

Table 1.1: Features of the CFM56-7B24 Engine (Reference 2)

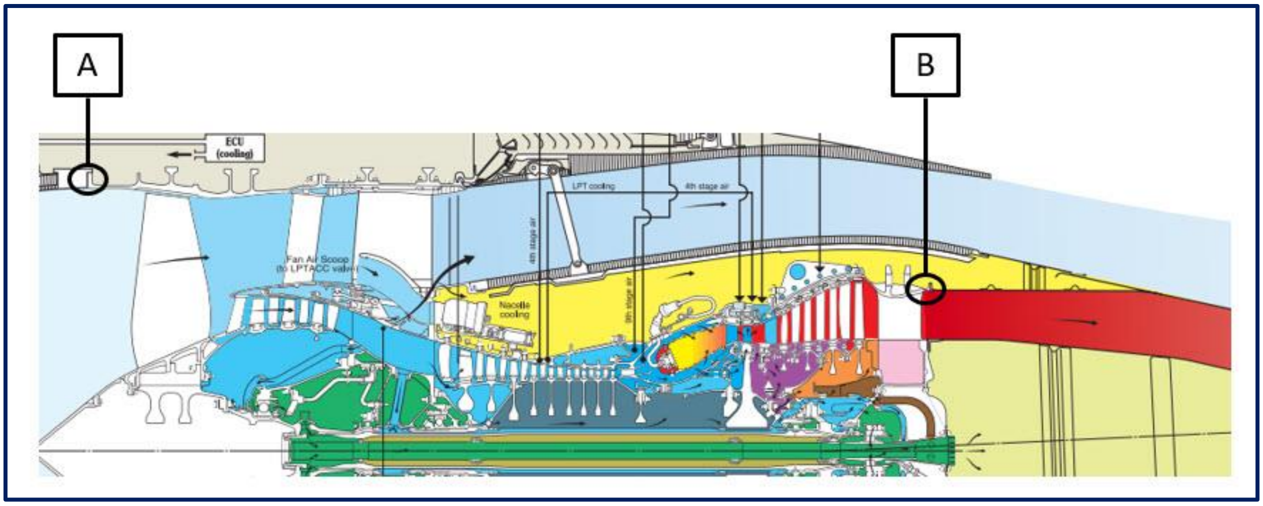

Figure 1.3: CFM56-7B24 Cross-Section with "Flange-to-Flange" Measurement Location

1.3 A Hybrid-Electric Propulsion System for a Commercial Transport

The study of various forms of electrically propelled aircraft has become increasingly important in the quest for lower consumption of carbon-based fuels [*3*]. Some electrified aircraft programs have focused on totally electric systems that use batteries but these have been limited essentially to commuter, on-demand mobility and air taxi services, mainly because of the excessive weight of batteries and their current low power density. It is recognized that a significant impact on global emissions will not be felt until such engines are widely used in commercial jet fleets [*4*]. Currently, rather than being totally electric, the most promising concepts are a mixture of "conventional" gas turbines and complementary electric propulsors – systems referred to as *hybrid-electric* engines. If we make realistic assumptions about the efficiencies of electrical systems and, say, an electrically driven fan, the overall cruise SFC changes very little. In fact, once the additional complexity, weight and cost are accounted for, there appears to be little reason for pursuing a hybrid concept. The main benefit must come from a better integration with the aircraft - both location and function - because it is the enabler for other *benefit magnifiers*, such as boundary layer ingestion, blown flaps, etc. Therefore, in this RFP, we focus a combination of two conventional primary gas turbine engines used to drive an electric fan that ingests boundary layer air.

2. Design Objectives and Requirements

2.1 The New Propulsion System

A hybrid-electric propulsion system is to be designed for the *NASA STARC-ABL Aircraft*. It is to be based on two new conventional turbofan engines carried on pylons beneath the wings. Power is to be extracted equally from the primary engines to drive an electric fan, which rotates around the rear of the fuselage. The electric fan ingests a substantial portion of the annular boundary layer. *Figure 2.1* illustrates the installation of the electric fan in the NASA program [1] and contains typical diameters and a length. The fan hub/tip radius ratio is 0.2963, but the dimensions of your fan do not need to be the same. Just take a look at the exterior of a *Boeing 737- 800* or an *Airbus A320* but note that, in the *NASA STARC-ABL*, the elevators are located at the tip of the vertical stabilizer. so their wakes will not be ingested by the fan.

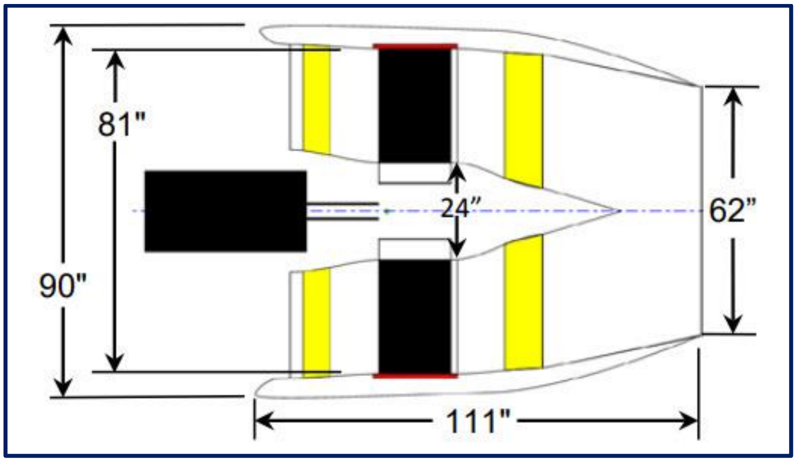

Figure 2.1: Assumed Geometry of Rear Fuselage [1]

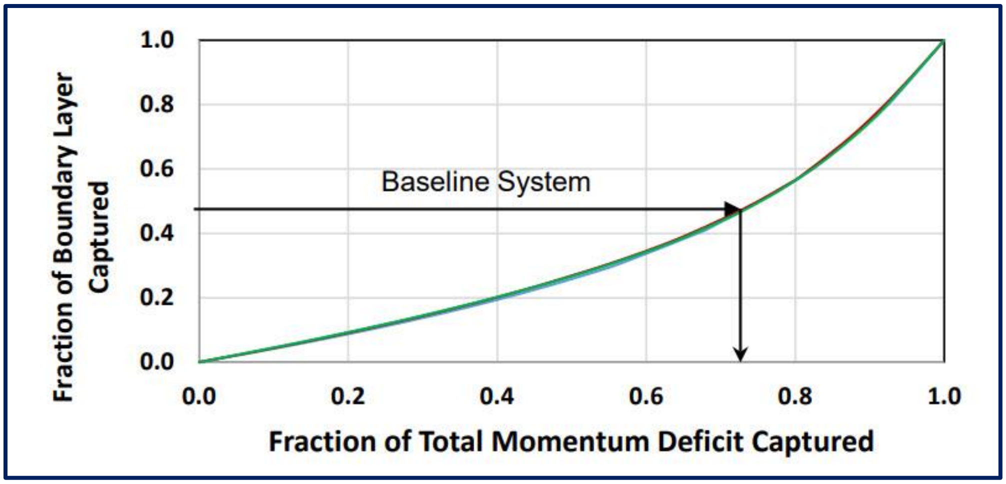

Figure 2.2: Boundary Layer Capture [1]

Figure 2.2 has also been taken from [1] and it reflects the observation that - for the NASA baseline case – roughly 72% of the boundary layer momentum is captured when roughly 48% of the boundary layer thickness is ingested. This provides us with an indication of the average velocity of the inlet flow of the electric fan when the aircraft is in motion.

2.2 General Objectives of the Engine Design Competition

The competition is intended to simulate a preliminary design project in industry. The objectives are

- To conduct a broad study of a matrix of engine designs using cycle and performance studies in order to determine how to focus the remainder of the new engine program.
- All candidate engines in your program should be designed to the same level by estimating the performance of individual major components and of the overall system. Their weights & dimensions should also be estimated, with the disks being sized with acceptable stress margins since they contribute substantially to the overall mass. The overall feasibility of each concept should be assessed; do they each fit together and operate as intended?
- Each of your candidate engines should be flown over a simple mission so that weight (more correctly mass) can be traded against performance and fuel burn. It is unlikely that the lightest propulsion system will consume the least fuel, so you will need to choose the best "compromised" solution to propose to your company as a candidate to be considered for more

detailed design work. The quality of your proposal in Round 1 will establish the confidence level for the investment of company resources.

- Round 2 of the competition serves as a design review by the Chief Engineer's Office, where the three most promising candidates will be ranked.

At this point, the budget is extremely tight and the risks are very high. No one is prepared to extend the exercise beyond 0-D (cycle studies) and 1-D (meanline studies). 2-D throughflow solutions are also unnecessary. Nothing is to be generated in 3 dimensions. Even though capabilities exist to produce elaborate 3-D assembly drawings, these are inappropriate because nothing will be designed in 3-D yet, and CFD is certainly not applicable. In the RFP, you are not being asked to demonstrate how much you know; you are being asked to apply only a certain amount of it and to focus that knowledge on the project in hand. The intention of the RFP is to provide a vehicle to help you learn and build confidence in applying important basic propulsion fundamentals.

Teams are limited to 4 people. This allows all team members to experience all aspects of the project fairly closely, while focusing on a specific part of it themselves – teamwork in action! To enable the project to be completed within a reasonable period, the project is deliberately restricted to preliminary design. If there are 6, 7 or 8 people who wish to participate, you have 2 teams! We can make an exception on team head count to accommodate an additional member. Just ask.

2.3 Some Specific Instructions

- Based on the entry-into-service date, which is 2035, development of new materials and an increase in design limits may be assumed.
- T4 may be increased to 3150 R.
    - Consider the use of carbon matrix composites in the HP turbine. Carefully justify your choices of any new materials, their location and the appropriate advances in design limits that they provide.
- T3 may be raised to 1620 R.
- Design proposals must include engine mass, engine dimensions, net thrust values, specific fuel consumption, thermal and propulsive efficiencies at cruise and take-off. Details of the major flow path components must be given. These include a simple parallel inlet (not the nacelle),

fan, booster, HP compressor, combustor, HP turbine, LP turbine, exhaust nozzle, bypass duct, and any inter-connecting ducts. Examples of velocity diagrams for only the turbines should be included to demonstrate their viability. This is not necessary for the compression system.

3. Baseline Engine Model

3.1 Take-Off Conditions: The Design Point

A generic model of the *CFM56-7B24* has been generated from publicly available information [2] using *GasTurb14* [5]. Details of this model are provided to assist with construction of your own baseline model to provide some indication of typical values of design parameters.

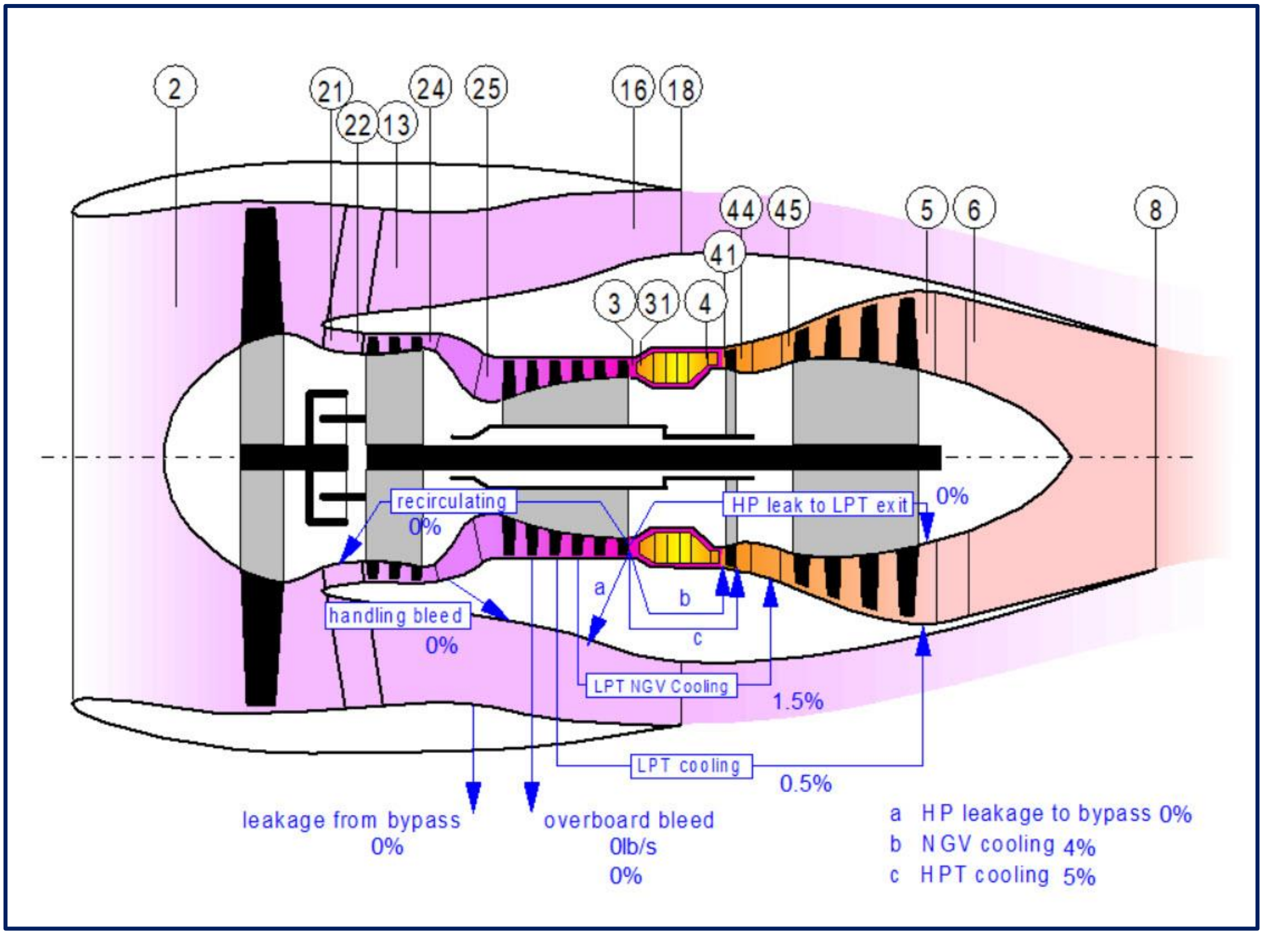

Figure 3.1: Turbofan Engine Schematic with Calculation Stations & Secondary Flows

Figure 3.1 contains a general schematic with relevant station numbers and secondary flow data for a non-augmented turbojet engine.

3.2 Overall Characteristics

Major Design Parameters

In a turbofan engine, four primary design variables are turbine entry temperature (*T4*), overall pressure ratio (*OPR* or *P3/P2*), bypass ratio and fan pressure ratio (*P21/P2*). For two spools the optimum energy division must be determined.

Property	*Unit*	*Value*	*Comment*
Intake Pressure Ratio		0.99	
No (0) or Average (1) Core dP/P		1	
Inner Fan Pressure Ratio		1.4	
Outer Fan Pressure Ratio		1.6	
Core Inlet Duct Press. Ratio		0.99	
IP Compressor Pressure Ratio		1.81	
Compr. Interduct Press. Ratio		0.98	
HP Compressor Pressure Ratio		10.5742	
Bypass Duct Pressure Ratio		0.98	
Turb. Interd. Ref. Press. Ratio		0.98	
Design Bypass Ratio		5.3	
Burner Exit Temperature	R	2800	
Burner Design Efficiency		0.9995	
Burner Partload Constant		1.6	used for off design only
Fuel Heating Value	BTU/lb	18552.4	
Overboard Bleed	lb/s	0	
Power Offtake	hp	150	
HP Spool Mechanical Efficiency		0.99	
Gear Ratio		1	
LP Spool Mechanical Efficiency		0.99	
Burner Pressure Ratio		0.96	
Turbine Exit Duct Press Ratio		0.96	

Table 3.1: Basic Cycle Input

Table 3.1 is the "Basic Input" for the design point of a *GasTurb14* model of the generic *CFM56-7B24* baseline. All four primary design variables are input, the overall pressure ratio being made up from the fan, the booster and the HPC, along with the inter-compressor duct loss. T4 was an estimated value. To generate an acceptable replica of the engine cycle, a unique combination of the remainder must be estimated iteratively using the net thrust (F_N) and specific fuel consumption (*sfc*) at design conditions as targets. By definition, this operating condition also corresponds to the entry points to any component performance maps, and this should be the case for your new engine.

The next four parameters relate to the primary combustor; they are all fairly conventional values by modern standards. The burner efficiency of 99.95% is current conventional value. A burner pressure loss of 4% is given up willingly to pay for complete mixing and efficient combustion, so this should be retained. The burner "*part load constant*" is an element in the calculation of burner efficiency discussed in the *GasTurb14 User Guide* [5]. Without expert knowledge, this is best left alone!

<u>Secondary Design Parameters</u>

Cooling Air: HPC air is bled from compressor delivery to cool the HP turbine vane and blade. Fully compressed air is an expensive commodity, but this is the only source that offers sufficient pressure to permit to coolant to be delivered to the hot vane and blade and emerge from their surfaces. This is aided by the pressure loss through the burner – another reason we can tolerate combustor pressure losses.

Property	*Unit*	*Value*	*Comment*
Rel. Handling Bleed to Bypass		0	
Rel. HP Leakage to Bypass		0	
Rel. Overboard Bleed W_Bld/W25		0	
Rel. Enthalpy of Overb. Bleed		1	
Recirculating Bleed W_reci/W25		0	Off Design Input Only
Rel. Enthalpy of Recirc Bleed		1	
Number of HP Turbine Stages		1	
HPT NGV 1 Cooling Air / W25		0.04	
HPT Rotor 1 Cooling Air / W25		0.05	
HPT Cooling Air Pumping Dia	in	0	Calculated in HPT Design
Number of LP Turbine Stages		4	
LPT NGV 1 Cooling Air / W25		0	
LPT Rotor 1 Cooling Air / W25		0.02	
LPT NGV 2 Cooling Air / W25		0	
LPT Rotor 2 Cooling Air / W25		0	
Rel. Enth. LPT NGV Cooling Air		0.6	
Rel. Enth. of LPT Cooling Air		0.7	
Rel. HP Leakage to LPT exit		0	
Rel. Fan Overb.Bleed W_Bld/W13		0	
Core-Byp Heat Transf Effectiven		0	
Coolg Air Cooling Effectiveness		0	
Bleed Air Cooling Effectiveness		0	

Table 3.2: Secondary Air System Input

Turbomachinery Efficiencies: Efficiency values may be entered directly via respective tabs on the input screen. Alternately, they may be calculated, based on aerodynamic and geometric data. Regardless of the input method, their values are given in *Table 3.4*. The designer has the choice of either isentropic or polytropic values, so he or she should be certain of their applicability and their definitions! However, another available option allows *GasTurb14* to calculate efficiencies from data supplied. Compressors utilize a NASA approach [6] but turbines first estimate prevailing values of stage loading and flow coefficients for use in a *Smith Chart* [7], assuming an equal work spilt between stages. This is a most convenient approach to turbine performance since various updated versions of the *Smith Chart* are available. More will be said about this topic in *Sub-sections 3.9* and *3.11*.

Power Off-take: All engines have power extracted - usually from the HP spool via a tower shaft that passes through an enlarged vane or strut in the main frame – to power aircraft systems. This is often preferred to the use of a separate auxiliary power unit, depending on how much power is required. We have selected a nominal power off-take of 150 hp from our baseline engine and this is indicated in the performance summary in *Table .3.4*. Modern engines tend to use a lot of this, so you might like to consider this issue for your engine and mission.

Dimensions: Diameters & Lengths: The engine cycle may be defined purely on the basis of thermodynamics. We define a "rubber engine" initially, where performance is delivered in terms of a net thrust at cruise - close to 24,200 lbf given in *Table 1.1* once the engine scale has been determined. For our baseline model, we also had a target dimensional envelope defined in Table *1.1*, namely a maximum fan diameter of 65 inches and a length of 98 inches. We have already discussed the merits of the latter. The diameter is determined from the mass flow rate and the Mach number at the fan face; the length is dealt with by manipulation of vane & blade aspect ratios and axial gaps in the turbomachinery and by suitable selection of duct lengths, usually defined as fractions of the corresponding entry radii. Once the correct thrust has been reached, the maximum radius is determined by setting an inlet radius ratio and then varying the Mach number at entry to the LPC. These values are input on the primary input screen under the LP compressor tab, where a Mach number of 0.58 was found to be appropriate - fairly low by today's standards – and is shown in *Table 3.7*. This sets the general radial dimension for the complete engine, although in fact downstream of the fan and booster, the entry radius of the HP compressor is also determined by input radius ratios and a value of local axial Mach number given in *Table 3.10.*

Name	*Where it is*	*Design Mach No*	*Design Area*
St2	Fan Inlet	Calculated by	LPC Design
St22	Booster Inlet	0.5	0
St24	Booster Exit	0.4	0
St25	HP Compressor Inlet	Calculated by	HPC Design
St3	HP Compressor Exit	0.25	0
St4	Burner Exit	0.1	0
St44	HP Turbine Exit	Calculated by	HPT Efficiency
St45	LP Turbine Inlet	0.35	0
St5	LP Turbine Exit	Calculated by	LPT Efficiency
St6	Exit Guide Vane Exit	0.45	0
St8	*Core Nozzle Throat*	*0*	*0*
St13	Bypass Inlet	0.55	0
St16	Bypass Exit	0.5	0
St18	*Bypass Nozzle Throat*	*0*	*0*

Table 3.3: Stations Input

The HP & LP turbine radii follow from the exit values of the respective upstream components. For the ducts, radial dimensions are keyed off the inner wall with the blade spans being superimposed. For the overall engine length, early adjustments are made by eye (My personal philosophy is that if it looks right, it's probably OK!), with final manipulations being added as the target dimension is approached. When modeling an existing engine, *GasTurb14* enables an available cross section to be located beneath the model, so that the model can be manipulated via numerical input or sliders assigned to input parameters, until a satisfactory match is achieved. The degree of success can be seen in *Figure 3.4*, where the cross section from *Figure 1.2* may be seen behind the model.

Materials & Weights: Use was made of the materials database in *GasTurb14*, where, in fact, the default selections were retained. For proprietary reasons, many advanced materials are not included. Examples of these are: polymeric composites used in cold parts of the engine, such as the inlet and fan; metal matrix composites, which might be expected in the exhaust system; carbon-carbon products, again intended for use in hot sections. All of these materials are considerably lighter than conventional alternatives, Within the component models, material densities can be modified independently of the database. While this was never implemented for our baseline, you may find it useful for your contemporary designs.

Component weights are calculated by multiplying the effective volumes by the corresponding material densities. Of course, only the major elements which are explicitly designed are weighed and there are many more constituents. Nuts, bolts, washers, seals and other much larger elements such as fuel lines, oil lines, pumps and control systems still must be accounted for. In industry, this is done by the application of a multiplier or adder to the predicted net mass, whose value is based on decades of experience, to obtain what is designated in the output as the total mass. In general, a multiplication factor of 1.3 is recommended in the *GasTurb14* manual, but we used a specific value of "*net mass factor*" in *Table 3.23* to reach the overall mass target.

Performance: A summary of the performance output for the generic *CFM56-7B24* model for the design point at static take-off is given in *Table 3.4*. The net thrust is within 1% of the target. The predicted specific fuel consumption of 1.36 is very close to the target value of 1.37 in *Figure 1.3*. See what you can produce in your baseline model!

A different format of thermodynamic output is contained in *Table 3.5*. Local values of mass flow rate, temperature, pressure, velocity, flow path area, axial Mach number, and radii - together with their axial locations - are especially useful.

Station	W lb/s	T R	P psia	WRstd lb/s
amb		518.67	14.696	
2	751.000	518.67	14.549	758.586
13	631.794	601.31	23.278	429.463
21	119.206	576.73	20.369	90.693
22	119.206	576.73	20.165	91.610
24	119.206	692.01	36.498	55.441
25	119.206	692.01	35.768	56.573
3	116.822	1394.29	378.223	7.442
31	106.094	1394.29	378.223	
4	108.541	2800.00	363.094	10.207
41	113.309	2745.66	363.094	10.552
43	113.309	2116.12	99.439	
44	119.270	2082.27	99.439	
45	121.058	2069.14	94.639	37.546
49	121.058	1550.17	25.745	
5	121.654	1548.49	25.745	119.989
8	121.654	1548.49	24.715	124.988
18	631.794	601.32	22.813	438.228
Bleed	0.000	1394.29	378.222	

Efficiency	isentr	polytr	RNI	P/P
Outer LPC	0.9000	0.9064	0.990	1.600
Inner LPC	0.9000	0.9047	0.990	1.400
IP Compressor	0.9208	0.9271	1.210	1.810
HP Compressor	0.9000	0.9260	1.727	10.574
Burner	0.9995			0.960
HP Turbine	0.8846	0.8679	3.526	3.651
LP Turbine	0.9218	0.9090	1.271	3.676

HP Spool mech Eff 0.9900 Nom Spd 14461 rpm
LP Spool mech Eff 0.9900 Nom Spd 5173 rpm

P22/P21=0.9900 P25/P24=0.9800 P45/P44=0.9517

hum [%]	war0	FHV	Fuel
0.0	0.00000	18552.4	Generic

FN	=	24227.50	lb
TSFC	=	0.3637	lb/(lb*h)
WF	=	2.44744	lb/s
s NOX	=	0.8197	
Core Eff	=	0.4506	
Prop Eff	=	0.0000	
BPR	=	5.3000	
P2/P1	=	0.9900	
P3/P2	=	26.00	
P5/P2	=	1.7695	
P16/P13	=	0.9800	
P16/P6	=	0.92304	
P16/P2	=	1.56800	
P6/P5	=	0.96000	
A8	=	375.29	in²
A18	=	1328.98	in²
XM8	=	0.90884	
XM18	=	0.81826	
WBld/W2	=	0.00000	
CD8	=	0.99248	
CD18	=	0.99099	
PWX	=	150.0	hp
V18/V8,id	=	0.57407	
WBLD/W22	=	0.00000	
Wreci/W25	=	0.00000	
Loading	=	100.00	%
WCHN/W25	=	0.04000	
WCHR/W25	=	0.05000	
WCLN/W25	=	0.01500	
WCLR/W25	=	0.00500	
WBLD/W25	=	0.00000	
WLkBy/W25	=	0.00000	
WlkLP/W25	=	0.00000	

Table 3.4: Baseline Engine Model Output Summary at Take Off

	Units	St 2	St 22	St 24	St 25	St 3	St 4	St 44	St 45	St 5	St 6	St 8	St 13	St 16	St 18
Mass Flow	lb/s	751	119.206	119.206	119.206	116.822	108.541	119.27	121.058	121.654	121.654	121.654	631.794	631.794	631.794
Total Temperature	R	518.67	576.727	692.01	692.01	1394.29	2800	2082.27	2069.14	1548.49	1548.49	1548.49	601.314	601.314	601.316
Static Temperature	R	485.923	549.369	670.683	665.234	1378.84	2795.91	2023.18	2030.26	1523.83	1497.56	1357.29	567.151	572.802	530.498
Total Pressure	psia	14.549	20.1649	36.4985	35.7685	378.223	363.094	99.4395	94.6395	25.7447	24.715	24.715	23.2784	22.8128	22.8128
Static Pressure	psia	11.5834	17.0011	32.6988	31.1399	362.606	360.755	88.0955	87.383	24.1496	21.6315	14.696	18.9559	19.2349	14.696
Velocity	ft/s	626.834	574.435	507.184	568.287	448.079	249.192	923.763	749.128	580.681	834.534	1609.36	641.917	586.43	923.892
Area	in²	2681.44	357.763	257.199	239.077	52.8933	180.101	158.196	200.311	705.277	538.424	372.467	1571.09	1711.68	1317.01
Mach Number		0.58	0.5	0.4	0.45	0.25	0.1	0.432339	0.35	0.310566	0.45	0.908836	0.55	0.5	0.818261
Density	lb/ft³	0.06434	0.083527	0.131591	0.126344	0.709794	0.348263	0.117527	0.11617	0.042775	0.038987	0.029224	0.090211	0.090636	0.07477
Spec Heat @ T	BTU/(lb*R)	0.240085	0.240577	0.241909	0.241909	0.261151	0.30316	0.288843	0.288391	0.273874	0.273874	0.273874	0.240861	0.240861	0.240861
Spec Heat @ Ts	BTU/(lb*R)	0.239981	0.240261	0.241663	0.2416	0.260663	0.3031	0.287463	0.287484	0.273076	0.272227	0.267507	0.240467	0.240532	0.240123
Enthalpy @ T	BTU/lb	-4.31602	9.65097	37.4383	37.4383	213.458	625.494	411.162	407.201	260.585	260.585	260.585	15.5774	15.5774	15.5778
Enthalpy @ Ts	BTU/lb	-12.1681	3.05675	32.2978	30.9845	209.446	624.253	394.108	395.986	253.847	246.668	208.826	7.34288	8.70492	-1.47999
Entropy Function @ T		-0.11924	0.252694	0.89265	0.89265	3.43965	6.44701	5.15293	5.1243	3.93608	3.93608	3.93608	0.399306	0.399306	0.399316
Entropy Function @ Ts		-0.34719	0.082028	0.782718	0.754073	3.39749	6.44055	5.0318	5.04453	3.87212	3.80282	3.41625	0.193897	0.228713	-0.040432
Exergy	BTU/lb	-0.357633	11.9898	38.1179	37.399	206.708	510.28	295.91	291.208	140.549	139.096	139.096	17.8084	17.0895	17.0896
Gas Constant	BTU/(lb*R)	0.068607	0.068607	0.068607	0.068607	0.068607	0.068606	0.068606	0.068606	0.068606	0.068606	0.068606	0.068607	0.068607	0.068607
Fuel-Air-Ratio		0	0	0	0	0	0.023069	0.02095	0.020634	0.020531	0.020531	0.020531	0	0	0
Water-Air-Ratio		0	0	0	0	0	0	0	0	0	0	0	0	0	0
Inner Radius	in	9.93653	14.6073	14.3152	8.31609	8.45126	12.8945	12.8945	12.8945	13.5392	13.5392	8.32142	19.4153	21.2094	21.2106
Outer Radius	in	30.8588	18.0901	16.9344	12.0523	9.39978	14.9531	14.7181	15.1667	20.1942	18.8334	13.737	29.615	31.5386	29.5451
Axial Position	in	22.1757	22.1757	46.065	58.0897	80.9547	92.2522	95.2796	96.5433	108.329	127.514	143.572	43.9119	69.9747	108.265

Table 3.5: Baseline Engine Model Detailed Output

A plot of the baseline engine model appears in *Figure 3.3* and as stated earlier, a comparison with the prototype cross section is shown in *Figure 3.4a*. *Figure 3.4b* is an over/under plot which compares the engine cross section with the model in a clearer manner. (You are requested to generate this type of plot of baseline versus new engine in your proposal.) Our inability to model neither the hot nor cold nozzles is apparent but the absolute accuracy of the baseline engine model in this exercise is of little consequence.

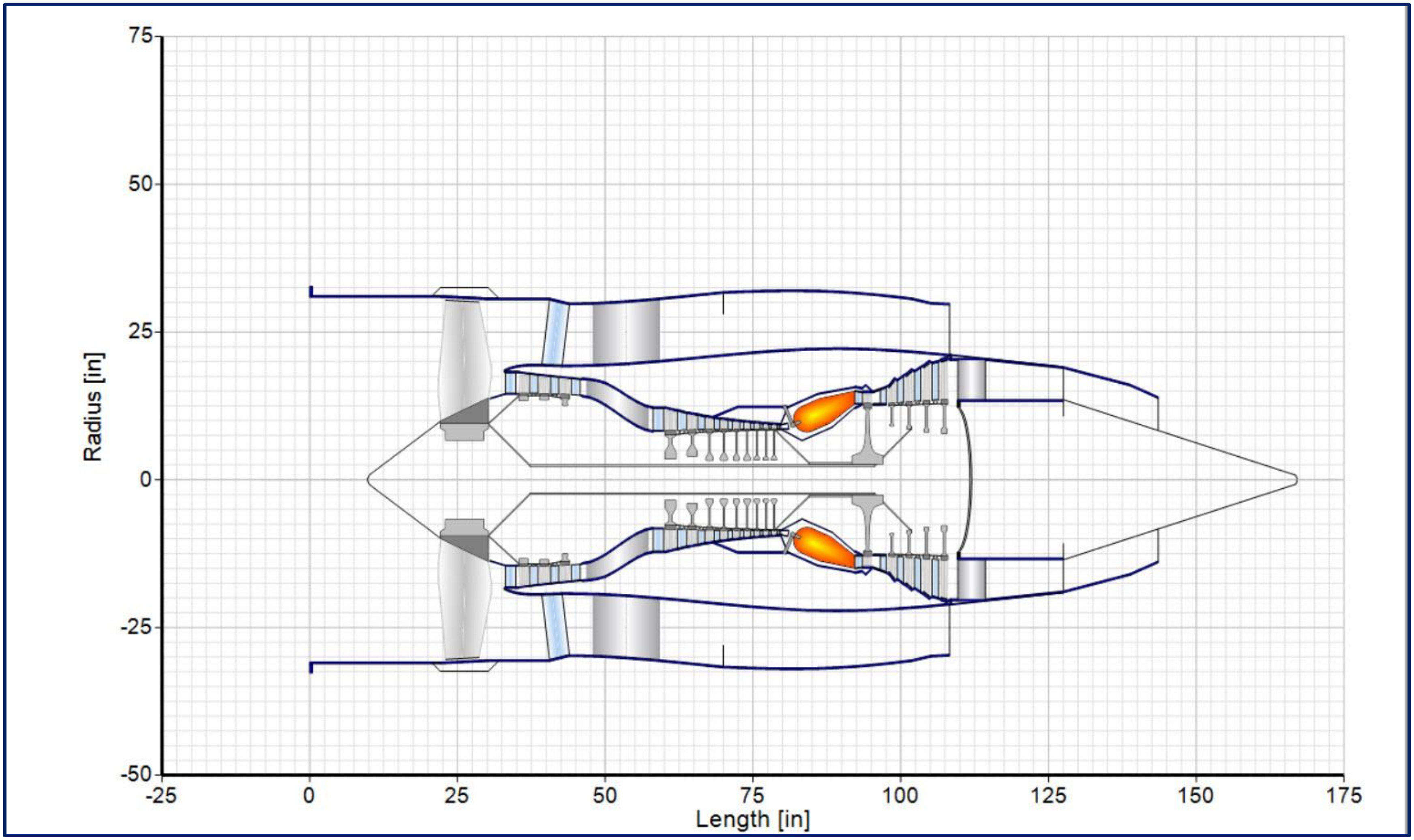

Figure 3.3: Baseline Engine Model Cross Section from GasTurb14

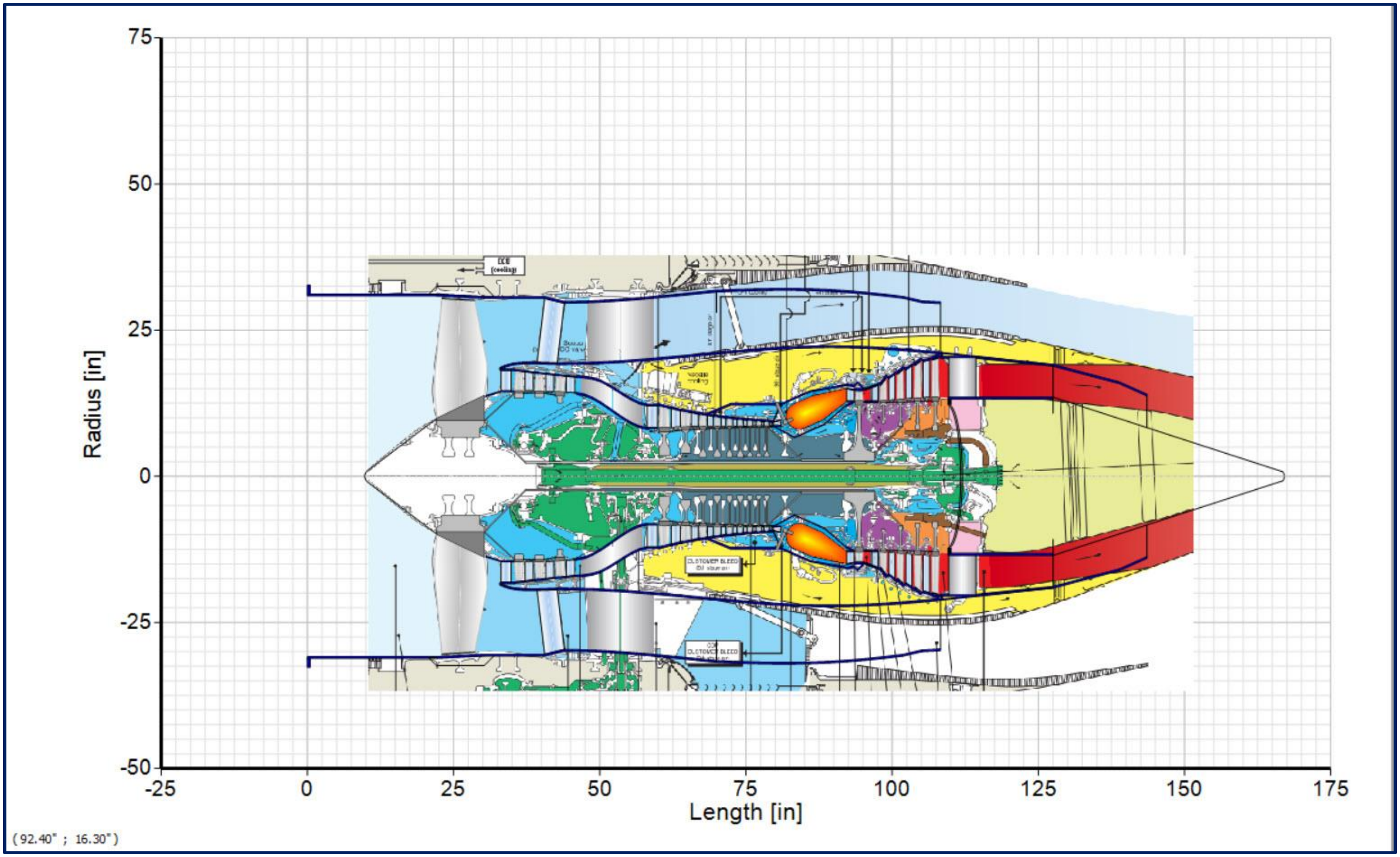

Figure 3.4a: Comparison of CFM56-7B24 Cross Section with GasTurb14 Baseline Model

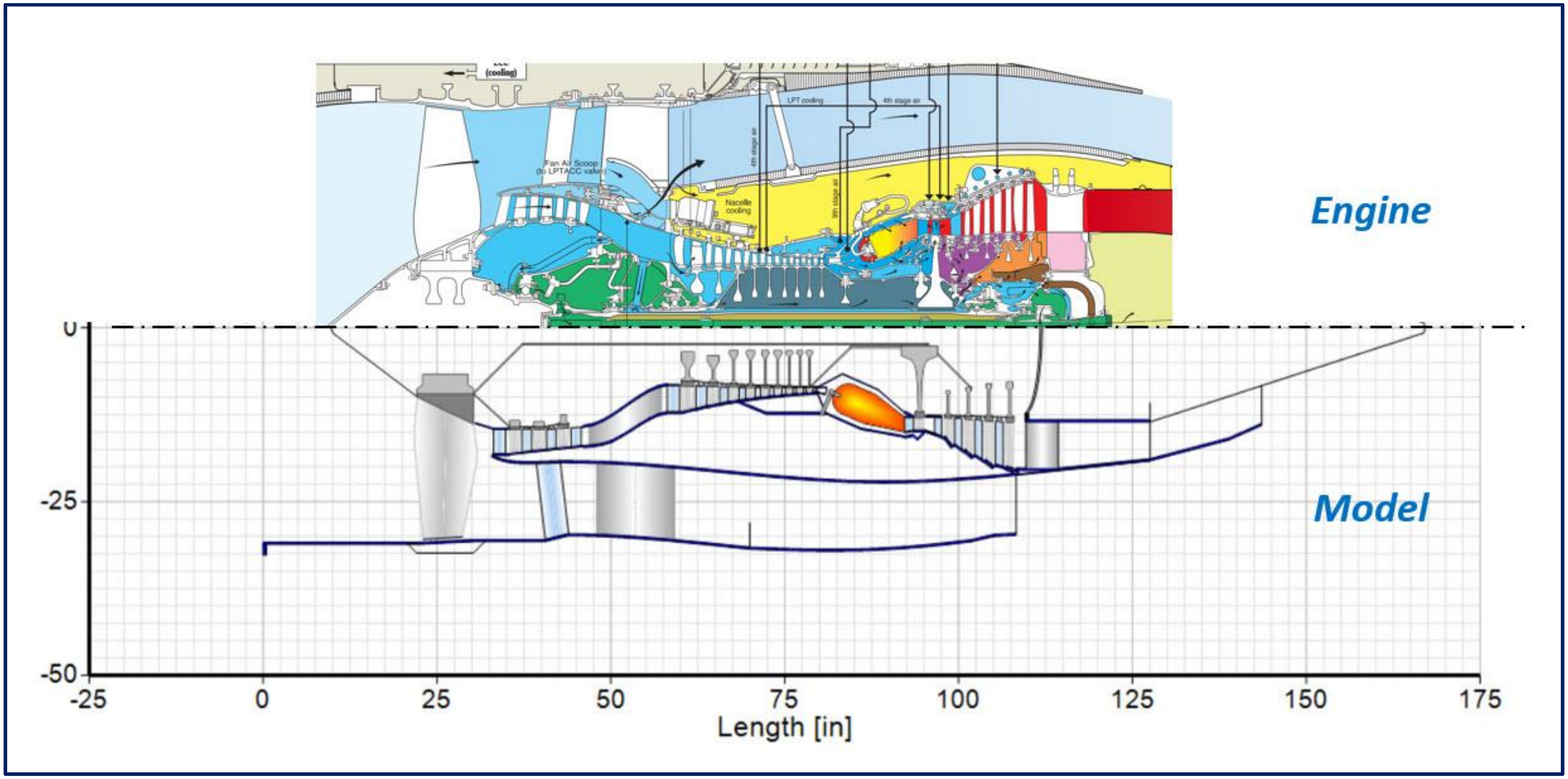

Figure 3.4b: Over/Under Comparison of CFM56-7B24 Cross Section with GasTurb14 Baseline Model

Some details of the component models now follow.

3.3 Inlet

Note that in this project we are not concerned with the real inlet and nacelle. We are currently interested in the hardware downstream of the inlet flange, as in *Figures 3.3* and *3.4*. The inlet is designed with an elliptical center body and the outer diameter and the inner shape of the inlet has been determined from those of the fan.

Number of Struts		0
Strut Chord/Height		0
Gap Width/Height		0.49
Cone Length/Radius		1.2
Cone Angle [deg]		37
Casing Length/Radius		0.25
Casing Thickness	in	0.3
Casing Material Density	lb/ft³	100
Inlet Mass Factor		1

Length	in	22.1757
Cone Length	in	11.9238
Cone Mass	lbm	8.92777
Casing Mass	lbm	74.6473
Strut Mass	lbm	0
Total Mass	lbm	83.5751

Table 3.6: Inlet Geometry Input & Output

Pertinent geometric characteristics are shown in *Table 3.6*. At 83.6 lbm, the inlet is fairly light and this is because, based on the density, we have taken a typical polymeric alloy as our choice of materials. This should accommodate the forces and any low dynamic heating effects of Mach 0.8 operation.

3.4 Fan

The fan characteristics are given in *Tables3.7*. The radius ratio and inlet Mach number are of particular interest because, when taken with mass flow rate, they define the fan are and tip radius. The rotational speed of the LP spool is set via the blade tip speed and tip radius. The value of corrected flow per unit area (40.7 lbm/ft^2) is fairly aggressive for a commercial engine and corresponds to the input value of Mach number 0.58. Your new design may exceed this.

Property	Unit	Value
LPC Tip Speed	ft/s	1393
LPC Inlet Radius Ratio		0.322
LPC Inlet Mach Number		0.58
Engine Inl/Fan Tip Diam Ratio		1
min LPC Inlet Hub Diameter	in	0

```
 Input:
LPC Tip Speed                    ft/s              1393.00
LPC Inlet Radius Ratio                             0.32200
LPC Inlet Mach Number                              0.58000
Engine Inl/Fan Tip Diam Ratio                      1.00000
min LPC Inlet Hub Diameter       in                0.00000
 Output:
LPC Tip circumf. Mach No                           1.28892
LPC Tip relative Mach No                           1.41341
Design LP Spool Speed       [RPM]                  5172.79
Design IP Spool Speed       [RPM]                  5172.79
LPC Inlet Tip Diameter           in               61.71755
LPC Inlet Hub Diameter           in               19.87305
Calculated LPC Radius Ratio                        0.32200
LP Spool Torque                  lb*ft            25394.73
Aerodynamic Interface Plane      in²               2991.63
Corr.Flow/Area LPC               lb/(s*ft²)       40.73791
```

Table 3.7: Fan Aerodynamics Input & Output

3.5 Booster

Number of Stages		3
Number of Variable Guide Vanes		0
Inlet Guide Vanes (IGV) 0/1		0
Annulus Shape Descr -0.5...1		-0.02
Blade and Vane Sweep 0/1		0
First Stage Aspect Ratio		2.4
Last Stage Aspect Ratio		1.9
Blade Gapping: Gap/Chord		0.24
Pitch/Chord Ratio		1
Disk Bore / Inner Inlet Radius		0.8
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	247.277
Rel Thickness Inner Air Seal		0.04
IP Compressor Mass Factor		1
Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	249.712
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*F	0.119503
Casing Time Constant		10
Blade and Vane Time Constant		0.5
Platform Time Constant		1
Design Tip Clearance [%]		1.5
d Flow / d Tip Clear.		2
d Eff / d Tip Clear.		2
d Surge Margin / d Tip Clear.		5

Length	in	10.6345
Total Number of Blade and Vanes		425
Casing Mass	lbm	33.4825
Total Vane Mass	lbm	22.1144
Total Blade Mass	lbm	40.5422
Inner Air Seal Mass	lbm	7.42054
Rotating Mass	lbm	97.6581
Total Mass	lbm	153.255
Polar Moment of Inertia	lb*in²	21714.7

Table 3.8: Booster Geometry Input & Output

3.6 Inter-Compressor Duct

Number of Struts		12
Length/Inlet Inner Radius		0.84
Inner Annulus Slope@Exit [deg]		-6
Relative Strut Length [%]		88
Casing Thickness	in	0.3
Casing Material Density	lb/ft³	249.712
Compr Interduct Mass Factor		1

Length	in	12.0247
Outer Casing Mass	lbm	51.2377
Strut Mass	lbm	34.9898
Inner Casing Mass	lbm	41.4204
Total Mass	lbm	127.648

Table 3.9: Inter-Compressor Duct Input & Output

Notice that in addition to using an overall net mass factor to adjust the engine weight, individual net mass factors may be applied to the components or net mass adders may be used. This remains at a value of unity for the inter-compressor duct at the bottom of the left-hand box in *Table 3.9* since little of the detailed structure, such as passage of service lines through the vanes and tower shaft for power extraction, is unaccounted for in our simple model.

3.7 High Pressure Compressor

```
 Input:
HPC Tip Speed                   ft/s            1521.00
HPC Inlet Radius Ratio                          0.69000
HPC Inlet Mach Number                           0.45000
min HPC Inlet Hub Diameter      in              0.00000
 Output:
HPC Tip circumf. Mach No                        1.20441
HPC Tip relative Mach No                        1.28573
Design HP Spool Speed      [RPM]               14461.45
HPC Inlet Tip Diameter          in             24.10460
HPC Inlet Hub Diameter          in             16.63217
Calculated HPC Radius Ratio                     0.69000
HP Spool Torque                 lb*ft          10856.89
Corr.Flow/Area HPC              lb/(s*ft²)     34.07463
```

Table 3.10: High Pressure Compressor Aerodynamics Input & Output

Again, we set the speed of the HP spool via the tip speed and the corresponding radius. General aerodynamic characteristics of the HP compressor are given in *Table 3.10*, while the geometry is defined in *Table 3.11*.

Number of Stages		9
Number of Radial Stages		0
Number of Variable Guide Vanes		0
Inlet Guide Vanes (IGV) 0/1		1
Annulus Shape Descriptor 0...1		0.09
Given Radius Rat: Inl/Exit 0/1		0
Inlet Radius Ratio		0.85
Exit Radius Ratio		0.93
Blade and Vane Sweep 0/1		0
First Stage Aspect Ratio		2.1
Last Stage Aspect Ratio		1.9
Blade Gapping: Gap/Chord		0.16
Pitch/Chord Ratio		1
Disk Bore / Inner Inlet Radius		0.3
Diffuser Area Ratio		1.5
Rel Work of Radial End Stage		0.3
Duct Inner Radius Ratio		1
Duct Length/Inlet Inner Radius		0
Number of Duct Struts		8
Relative Duct Strut Length [%]		60
Rad Diffusor/Rotor Blade Length		0.5
Rotor Inlet Swirl Angle		0
Rotor Blade Backsweep Angle		20
Diffusor Wall Thickness	in	0.09842
IGV Profile Thickness [%]		5
IGV Material Density	lb/ft³	249.712
Rel Thickness Inner Air Seal		0.04
Compressor Mass Factor		1
Outer Casing Thickness	in	0.3
Outer Casing Material Density	lb/ft³	283.4
Casing Thickness	in	0.3
Casing Material Density	lb/ft³	283.4
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*F	0.11950
Casing Time Constant		10

Length (w/o Diffusor)	in	21.5775
Number of Inlet Guide Vanes		36
Total Number of Blade and Vanes		1671
Diffusor Length	in	1.28752
Casing Mass	lbm	71.6207
Outer Casing Mass	lbm	47.929
Total Vane Mass	lbm	14.4303
Total Blade Mass	lbm	34.349
Inner Air Seal Mass	lbm	6.53455
Rotating Mass	lbm	139.436
IGV Mass	lbm	3.07627
Exit Diffusor Mass	lbm	7.15851
Total Mass	lbm	283.651
Polar Moment of Inertia	lb*in²	7103.06

Table 3.11: High Pressure Compressor Geometry Input & Output

3.8 Combustor

Reverse Flow Design (0/1)		0
Outer Casing Length/Length		1.7
Exit/Inlet Radius		1.56
Length/Inlet Radius		1.41
Can Width/Can Length		0.4
Inner Casing Thickness	in	0.0787402
Outer Casing Thickness	in	0.19685
Casing Material Density	lb/ft³	499.424
Can Wall Thickness	in	0.3
Can Material Density	lb/ft³	499.424
Can Thermal Exp Coeff	E-6/R	18
Can Specific Heat	BTU/(lb*F	0.119503
Can Time Constant		1
Mass of Fuel Inj. / Fuel Flow		2
Burner Mass Factor		1

Mean Radius, Exit	in	13.9238
Length	in	12.585
Can Volume	in³	2960.58
Can Mass	lbm	156.543
Can Surface Area / Mass	in²/lbm	23.0666
Fuel Injector Mass	lbm	4.89488
Inner Casing Mass	lbm	19.3436
Outer Casing Mass	lbm	71.7238
Total Mass	lbm	252.506
Can Heat Soakage	hp	0

Table 3.12: Combustor Geometry Input & Output

A fairly conventional annular combustor is used and geometric details are given in *Table 3.12*. The high density of its material corresponds to the necessary thermal properties. The combustor is a major structural component, linked closely to the HP turbine first vane assembly. This is emphasized by its significant mass.

3.9 High-Pressure Turbine

Property	*Unit*	*Value*
1. HPT Rotor Inlet Dia	in	19.685
Last HPT Rotor Exit Dia	in	20.8661
HPT Exit Radius Ratio		0.77
HPT Vax.exit / Vax.average		1
HPT Loss Factor [0.3...0.4]		0.35
HPT 1. Rotor Cooling Constant		0.05
Interduct Reference Mach No.		0.4

Table 3.13: Input to Calculate High Pressure Turbine Efficiency

We chose to have *GasTurb14* calculate isentropic efficiency based on the data shown in *Table 3.13*, because additional valuable information is then generated, in addition to velocity diagrams and the corresponding *Smith Chart* [7]. Note that the values of the efficiency contours are expressed as fractions of the maximum value on the chart.

A general summary of the HP turbine aerodynamics and performance is presented in *Table 3.14*, followed by the velocity diagrams and *Smith Chart* in *Figure 3.5*. In *Table 3.14*, the value of AN^2 (a measure of the disk rim stress) at almost 37 x 10^9 $in^2 rpm^2$, is fairly modest by today's standards high compared with a typical limit value of 45 x 10^9. That informs us that higher rotational speeds are feasible in your new engine designs – depending on the geometry! In contrast, the velocity diagram in *Figure 3.5*, is fairly aggressive, with a high blade turning angle around 120° degrees and a stage loading coefficient $\psi = \Delta H/U^2$ of 2.69. What the *Smith Chart* tells us is that we may be able to reduce the stage loading coefficient to a value near 1.7 by increasing the mean blade speed via a higher mean radius and inlet radius ratio. But if we were to do that, the stage flow coefficient $\varphi = Va/U$ would need to be held constant by squeezing the flow are to increase the axial flow velocity. The efficiency would improve as the HPT design point moved vertically downwards. The trade-off against mass would then need to be considered in the final comparison of engine candidates.

Input:			
Number of Stages			1
Last HPT Rotor Exit Dia	in		20.86614
HPT Exit Radius Ratio			0.77000
HPT Vax.exit / Vax.average			1.00000
HPT Loss Factor [0.3...0.4]			0.35000
HPT 1. Rotor Cooling Constant			0.05000
Interduct Reference Mach No.			0.40000
Output:			
HPT Inlet Radius Ratio			0.92518
HPT First Stator Exit Angle			69.51387
HPT Exit Mach Number			0.68435
HPT Exit Angle			-50.82145
HPT Last Rotor abs Inl Temp	R		2745.66
HPT First Rotor rel Inl Temp	R		2430.22
HPT First Stage H/T	BTU/(lb*R)		0.06796
HPT First Stage Loading			2.69304
HPT First Stage Vax/u			0.68988
HPT Exit Tip Speed	ft/s		1487.74
HPT Exit A*N*N	in^2*RPM^2*E-6		37171.71
HPT 1.Rotor Cool.Effectiveness			0.50000
HPT 1.Rotor Bld Metal Temp	R		1912.26
Velocities:			
Stage Inlet Absolute Velocity	V	ft/s	2595.37
Stage Inlet Axial Velocity	Vax	ft/s	908.33
Stage Inlet Relative Velocity	W	ft/s	1437.82
Circumferential Velocity	U	ft/s	1316.65
Stage Exit Absolute Velocity	V	ft/s	1437.82
Stage Exit Axial Velocity	Vax	ft/s	908.33
Stage Exit Relative Velocity	W	ft/s	2595.37
Warning:			
Last Rotor Exit Mean Dia is not consistent with annulus in Station St44			

Table 3.14: High Pressure Turbine Aerodynamics Output

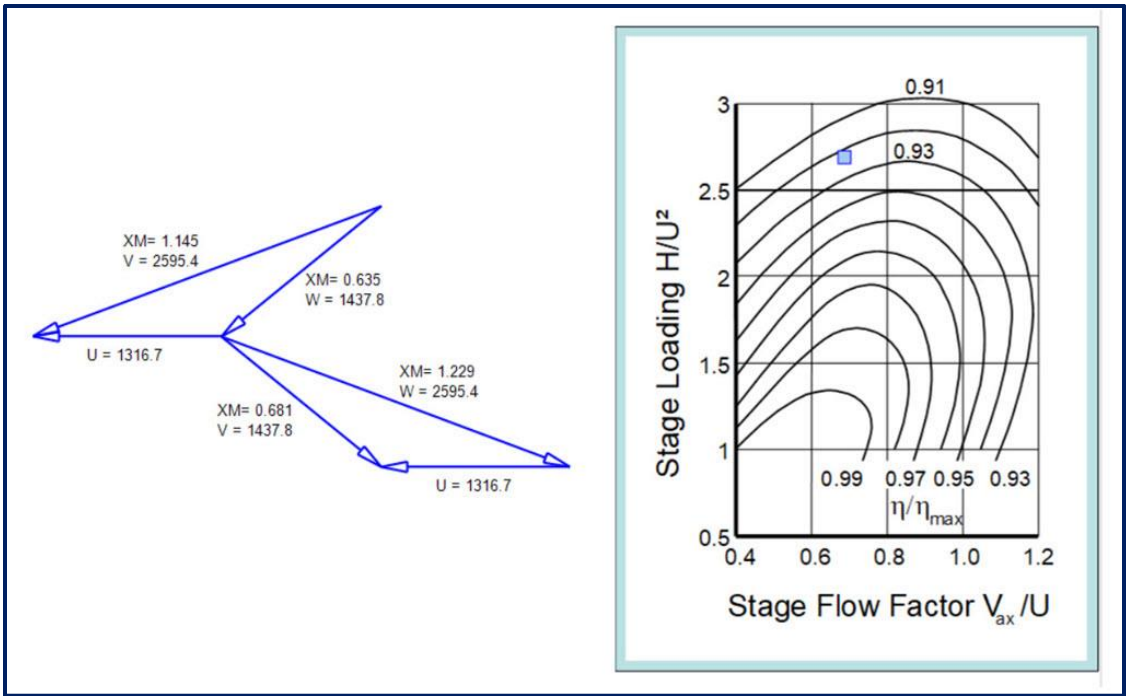

Figure 3.5: High Pressure Turbine Velocity Diagram & Smith Chart

Number of Stages = 1		no input
Unshrouded/Shrouded Blades 0/1		0
Inner Radius: R,exit / R,inlet		1
Inner Annulus Slope @Inlet[deg]		0
Inner Annulus Slope @Exit [deg]		-5
First Stage Aspect Ratio		1.7
Last Stage Aspect Ratio		1.3
Blade Gapping: Gap/Chord		0.25
Pitch/Chord Ratio		1
Disk Bore / Inner Inlet Radius		0.2
Rel Thickness Inner Air Seal		0.04
HP Turbine Mass Factor		1
Outer Casing Thickness	in	0.3
Outer Casing Material Density	lb/ft³	499.424
Casing Thickness	in	0.3
Casing Cooling Effectiveness		0.5
Casing Material Density	lb/ft³	499.424
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*F	0.119503
Casing Time Constant		20
Blade and Vane Time Constant		2
Platform Time Constant		5
Design Tip Clearance [%]		1.5
d Eff / d Tip Clear.		2

Length	in	3.02739
Total Number of Blade and Vanes		144
Casing Mass	lbm	24.5417
Outer Casing Mass	lbm	29.6971
Total Vane Mass	lbm	6.89477
Total Blade Mass	lbm	18.2465
Inner Air Seal Mass	lbm	0
Rotating Mass	lbm	152.39
Total Mass	lbm	213.524
Polar Moment of Inertia	lb*in²	10350

Table 3.15: High Pressure Turbine Geometry Input & Output

HP turbine geometric details are shown in *Table 3.15*.

We have already emphasized the critical role of disk weights in a practical engine model, so with that in mind, let us illustrate how disk sizing is carried out, using the single HPT stage as an example. *Table 3.16*, from *GasTurb14*, illustrates this. The three sections of the figure show the input, the boundary conditions and the output as we move from left to right.

In the input, a realistic radial temperature has not replaced the default value of 300F, since this is used in estimating transient behavior. We have selected a hyperbolic disk and set stress margins. The major geometrical controls for the disk design appear in the lower portion of the input table. The lower the bore radius the better, since radial stresses are reduced. A satisfactory disk solution is brought about by a smooth manipulation of the available features, usually one at a time! *GasTurb14* permits the search for suitable combinations to be done automatically but I prefer the old fashioned manual method, because then I can see what is happening!

The boundary conditions, in the central display, include features and conditions from the flowpath and the blade count. There is a default for the blade and vane solidity, which is normally set to 0.5. I find very frequently that this results in an excessive number of rotors, which leads to difficulties in meeting the disk stress limits. The blade count is altered by manipulating the Pitch/Chord Ratio in Table 3.15. A value of unity reduces the number of blades by 50%.

In the output display, look for a positive value of *Minimum Margin (%)*. A value in excess of zero results in aa disk that is acceptable but something around 10 to 20% is better because it offers a more stable solution and also one which is acceptable at overspeed off-design conditions. (See the comment in the second bullet in *Sub-section 4.1*.)

		Stage 1
Temperature Gradient	F	360
Temperature Adder	F	0
Web/Hyperbolic Disk(1/2)		2
Adapt Bore Width (0/1)		0
Adapt Bore Radius (0/1)		0
Optimize Disk (0/1)		0
Design Stress Margin [%]		20
Design Burst Margin [%]		20
Design Bore Stress Margin [%]		20
Design Web Stress Margin [%]		20
Design Burst Speed [%]		130
Blade Material Density	lb/ft^3	499.42
Blade Elasticity	ksi	21756
Blade Thermal Exp Coeff	E-6/R	18
Blade Specific Heat	BTU/(lb*	0.1195
Mean Bld Thickness,[%] of Chor		9
Root Height/Blade Height		0.4
Inner Rim Angle [°]		60
Web Width/Rim Width		0.26
Inner Rim Height/Rim Width		1.1
Bore Width Input	in	5.22
Bore Radius Input	in	2.6401

		Stage 1
Design Speed [RPM]		14461
Mean Line Position	in	94.371
Blade Inlet Root Radius	in	12.921
Blade Exit Root Radius	in	12.895
Casing Rotor Inlet Radius	in	14.718
Casing Rotor Exit Radius	in	14.718
Rim Width (= Axial Chord)	in	1.211
Blade Annulus Height	in	1.8106
Length of Blade	in	1.7834
Number of Blades		72
Unshrouded/Shrouded (0/1)		0
Number of Vanes		72
Single Vane Surface Area	in^2	2.4929
Rim Temperature Base Value	F	8166.8
Casing Temperature	F	1473
Blade Temperature	F	2011.4
Platform Temperature	F	1764.3
Gas Temperature	F	2011.4

		Stage 1
Rim Radius, Live Disk	in	12.285
Actual Bore Radius	in	2.6632
Actual Bore Width	in	5.22
Rim Load	ksi	21.159
Disk Mass incl Posts	lbm	134.14
Blade Mass	lbm	18.247
Inertia - Live Disk	lb*in^2	6103.1
Inertia - Total	lb*in^2	10350
Rim Temperature	F	490.23
Bore Temperature	F	648.25
Average Temperature	F	744.69
Average Tangential Stress	ksi	75.055
Stress Margin [%]		40.072
Burst Margin [%]		20.815
Bore Stress Margin [%]		40.072
Web Stress Margin [%]		68.753
Burst Speed [%]		160.33
Minimum Margin [%]		0.81512
Overstressed (0/1)		0
Platform Radius	in	12.946
Blade Length	in	1.7835
Blade Tip Radius	in	14.73
Casing Radius	in	14.216
Tip Clearance [%]		-40.527
Delta Tip Clear. Transient [%]		0
Heat Soakage	BTU/s	0
Casing Temperature	F	1473
Blade&Vane Area/Mass	in^2/lbm	52.065
Platform Area / Mass	in^2/lbm	15.626
Casing Area / Mass	in^2/lbm	9.1261

Table 3.16: High Pressure Turbine Disk Input, Boundary Conditions & Output

3.10 Inter-Turbine Duct

Number of Struts		0
Exit/Inlet Inner Radius		1
Length/Inlet Inner Radius		0.098
Inner Annulus Slope@Inlet[deg]		0
Inner Annulus Slope@Exit [deg]		0
Relative Strut Length [%]		0
Casing Thickness	in	0.3
Casing Material Density	lb/ft^3	499.424
Turbine Interduct Mass Factor		1

Length	in	1.26366
Outer Casing Mass	lbm	10.9157
Strut Mass	lbm	0
Inner Casing Mass	lbm	8.87691
Total Mass	lbm	19.7926

Table 3.17: Inter-Turbine Duct Input & Output

3.11 Low-Pressure Turbine

Characteristics of the low pressure turbine are presented in *Tables 3.18 to* 3.20 and *Figure 3.6*. Except for the comments about excessive disk rim stress, the discussion is the same as for the HP turbine.

Property	*Unit*	*Value*
LPT with EGV's [0/1]		1
1. LPT Rotor Inlet Dia	in	35.4331
Last LPT Rotor Exit Dia	in	45
LPT Exit Radius Ratio		0.8
LPT Vax.exit / Vax.average		1
LPT Loss Factor [0.3...0.4]		0.35
LPT 1. Rotor Cooling Constant		0

Table 3.18: Input to Calculate Low Pressure Turbine Efficiency

```
 Input:
Number of Stages                                  4
LPT with EGV's [0/1]                              1.00000
1. LPT Rotor Inlet Dia          in               35.43307
Last LPT Rotor Exit Dia         in               45.00000
LPT Exit Radius Ratio                             0.80000
LPT Vax.exit / Vax.average                        1.00000
LPT Loss Factor [0.3...0.4]                       0.35000
LPT 1. Rotor Cooling Constant                     0.00000
 Output:
LPT Inlet Radius Ratio                            0.85275
LPT First Stator Exit Angle                      61.55078
LPT Exit Mach Number                              0.31178
LPT Exit Angle                                   -5.05453
LPT Last Rotor abs Inl Temp     R                 1710.96
LPT First Rotor rel Inl Temp    R                 2019.12
LPT First Stage H/T             BTU/(lb*R)        0.01766
LPT First Stage Loading                           1.42905
LPT First Stage Vax/u                             0.56908
LPT Exit Tip Speed              ft/s              1128.53
LPT Exit A*N*N                  in²*RPM²*E-6     18913.96
LPT 1.Rotor Cool. Effectiveness                   0.00000
LPT 1.Rotor Bld Metal Temp      R                 2019.12
LPT Torque                      lb*ft            25394.73

 Velocities:
1st  Stage Inlet Absolute Velocity  V     ft/s     955.37
1st  Stage Inlet Axial Velocity    Vax    ft/s     455.12
1st  Stage Inlet Relative Velocity  W     ft/s     456.90
1st  Circumferential Velocity       U     ft/s     799.74
1st  Stage Exit Absolute Velocity   V     ft/s     456.90
1st  Stage Exit Axial Velocity     Vax    ft/s     455.12
1st  Stage Exit Relative Velocity   W     ft/s     955.37

Last Stage Inlet Absolute Velocity  V     ft/s    1213.32
Last Stage Inlet Axial Velocity    Vax    ft/s     578.00
Last Stage Inlet Relative Velocity  W     ft/s     580.26
Last Circumferential Velocity       U     ft/s    1015.68
Last Stage Exit Absolute Velocity   V     ft/s     580.26
Last Stage Exit Axial Velocity     Vax    ft/s     578.00
Last Stage Exit Relative Velocity   W     ft/s    1213.32

 Warning:
1. Rotor Inlet Mean Dia is not consistent with annulus in Station St45

 Warning:
Last Rotor Exit Mean Dia is not consistent with annulus in Station St5
```

Table 3.19: Low Pressure Turbine Aerodynamics Input & Output

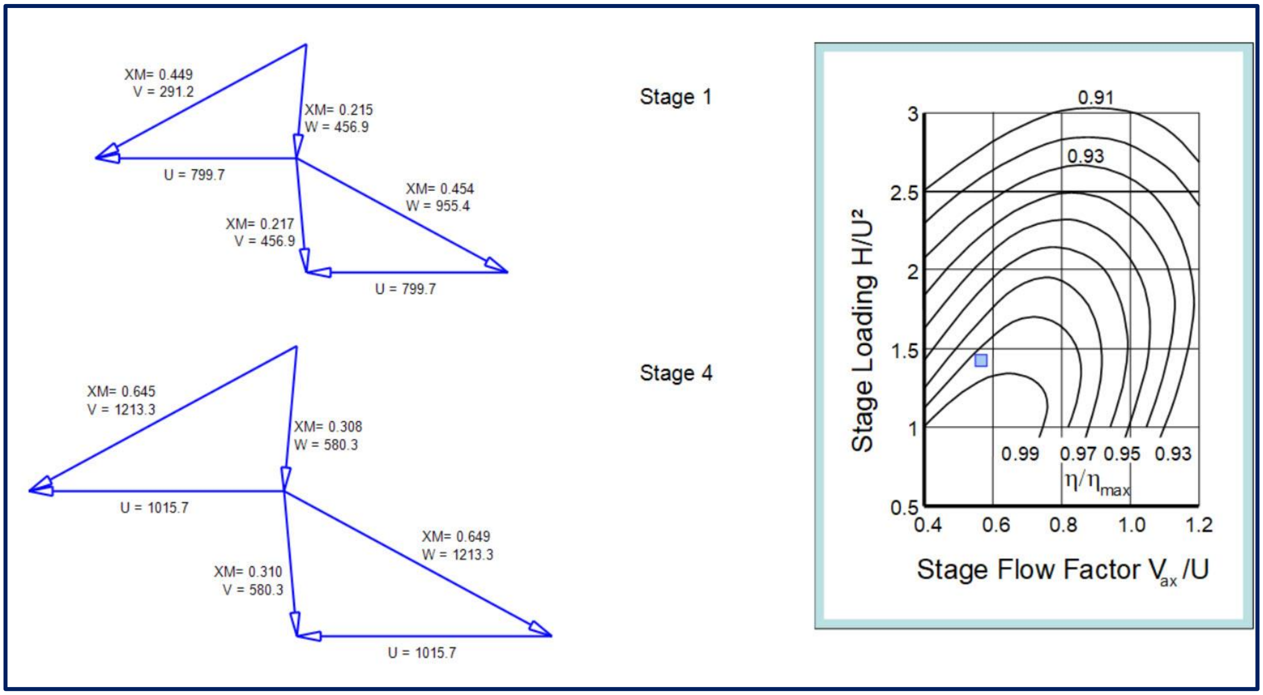

Figure 3.6: Low Pressure Turbine Velocity Diagram & Smith Chart

The *Smith Chart* in *Figure 3.6* indicates that our LPT has a better isentropic efficiency than its HP counterpart and this is confirmed by the less aggressive velocity diagrams for the first and last stages.

Number of Stages = 4		no input
Unshrouded/Shrouded Blades 0/1		1
Inner Radius: R,exit / R,inlet		1.05
Inner Annulus Slope@Inlet[deg]		0
Inner Annulus Slope@Exit [deg]		0
First Stage Aspect Ratio		2.23
Last Stage Aspect Ratio		2.1
Blade Gapping: Gap/Chord		0.4
Pitch/Chord Ratio		0.75
Disk Bore / Inner Inlet Radius		0.6
Rel Thickness Inner Air Seal		0.04
LP Turbine Mass Factor		1
Casing Thickness	in	0.3
Casing Cooling Effectiveness		0.5
Casing Material Density	lb/ft³	499.424
Casing Thermal Exp Coeff	E-6/R	18
Casing Specific Heat	BTU/(lb*F	0.119503
Casing Time Constant		20
Blade and Vane Time Constant		2
Platform Time Constant		5
Design Tip Clearance [%]		1.5
d Eff / d Tip Clear.		2

Length	in	11.7858
Total Number of Blade and Vanes		956
Casing Mass	lbm	123.419
Total Vane Mass	lbm	85.2115
Total Blade Mass	lbm	154.757
Inner Air Seal Mass	lbm	2.1335
Rotating Mass	lbm	283.559
Total Mass	lbm	492.189
Polar Moment of Inertia	lb*in²	54156.4

Table 3.20: Low Pressure Turbine Geometry Input & Output

3.12 Exhaust and Nozzle

The core exhaust is directly downstream of the low pressure turbine. It is comprised of an outer & inner casing, and a cone that closes off the inner casing. There is also a set of struts - a frame, which supports the rear bearing and centers the rotating assembly. *Table 3.21* contains the input and output details of the exhaust geometry.

Number of Struts		8
Strut Chord/Height		0.7
Strut Lean Angle		0
Gap Width/Height		0.2
Cone Angle [deg]		0
Cone Length/Inlet Radius		1
Casing Length/Inlet Radius		0.95
Bypass Radius/Flange Radius		1
Inner Casing Thickness	in	0.3
Outer Casing Thickness	in	0.3
Casing Material Density	lb/ft³	499.424
Exhaust Duct Mass Factor		1

Length	in	19.1845
Cone Length	in	13.195
Outer Casing Mass	lbm	192.269
Strut Mass	lbm	43.0093
Cone Mass	lbm	165.881
Front Cover Mass	lbm	44.9976
Total Mass	lbm	446.157

Table 3.21: Exhaust Geometry Input & Output

Geometry and mass of the core nozzle elements are presented in *Table 3.22*.

Std/Plug/Power Gen Exh 1/2/3		2
Inl Section Length/Outer Radius		0.6
Conv Length/Inl Section Radius		0.3
Cone Angle [deg]		18
Cone Length/Inlet Radius		2.9
Inlet Section Area Ratio		0.9
Inner Casing Thickness	in	0.3
Outer Casing Thickness	in	0.3
Casing Material Density	lb/ft³	499.424
Nozzle Mass Factor		1

Overall Length	in	39.2638
Inlet Section Length	in	11.3
Convergent Length	in	4.75873
Convergent Cone Angle [deg]		24.0679
Inlet Section Mass	lbm	110.425
Convergent Section Mass	lbm	42.0214
Inner Casing Mass	lbm	161.161
Outer Casing Mass	lbm	152.446
Total Mass	lbm	313.607

Table 3.22: Nozzle Geometry Input & Output

3.13 Overall Engine

LP Shaft Thickness	in	0.19685
HP Shaft Thickness	in	0.19685
Shaft Material Density	lb/ft³	499.424
LP Spool Design Spd Incr [%]		0
HP Spool Design Spd Incr [%]		0
Gear Box Mass / Power	lbm/hp	0
Net Mass Factor		1.2267
Net Mass Adder	lbm	0

Front LP Shaft Cone Length	in	7.06786
Middle LP Shaft Length	in	58.2837
Middle LP Shaft Radius	in	2.46632
Rear LP Shaft Cone Length	in	6.12936
HP Shaft Cone Length	in	5.80072
HP Shaft Length	in	7.2221
HP Shaft Radius	in	2.66317
Engine Length	in	166.777
Max Engine Diameter	in	68.2462
Nacelle Length (Bypass only)	in	108.265
LP Shaft Mass	lbm	80.3185
HP Shaft Mass	lbm	19.1118
Gear Box Mass	lbm	0
Net Mass	lbm	4260.21
Total Mass	lbm	5226
LP Spool Inertia	lb*in²	179078
HP Spool Inertia	lb*in²	17453

Table 3.23: Overall Engine Input & Output

The dry weight of 5432 lbm, published in *Reference 2*, excludes the nozzle, so strictly we should allow for this in *Table 3.23* when we estimate the net mass factor that accounts for the secondary systems outside the flow path that are not included in our preliminary engine design. When we do this, however, it makes very little difference to the predicted value of the overall engine mass. So we ignored its effect. Geometric details of the overall engine are provided in *Table 3.23*. The overall total mass of the engine, at 5226 is 0.15% less than our target value. Good enough, we have more important things to worry about!

4. The New Propulsion System; Specific Instructions

4.1 Overall Approach to the RFP

- The task of building the baseline engine model is deliberately set to provide training and experience in generating a model that works and looks right. Essentially, you are addressing a problem to which you have been given the answer. The baseline model data in *Section 3* contains typical values for a multitude of parameters and thermodynamic stations. It exemplifies what you should include in your proposal in quality and technical level. The design point for your baseline engine model is at static sea level take-off conditions. You should replicate the baseline engine model with whatever software you will use for your new engine design so you have your own version of the baseline. Describe briefly how your baseline model was generated. This is an important learning exercise! Your results may not match our baseline model exactly but is essential for you to make a valid comparison of weights and performance against your new hybrid-electric propulsion system candidates.

- When we ran our baseline turbofan model off-design at TOC (0.8 Mach, 35,000 ft altitude), the LP spool speed increased from 5273 rpm to 5366 rpm. This meant that the LPT disks no longer were within their stress margins. Therefore it is recommended that you “over-design” your disks in your baseline model at static take-off.

- The new hybrid-electric propulsion system is to be designed at top-of-climb (TOC) or cruise conditions. This is because engines for commercial passenger aircraft are usually designed where most of the fuel is consumed. Unfortunately, very little information at cruise conditions is made public at and we have been forced to work with what we could get. So you must run your baseline engine model off-design at an altitude of 35,000 ft. and Mach 0.8 to determine the *STARC-ABL* aircraft thrust requirement at cruise with two baseline engines. This is the overall target net thrust for your new hybrid-electric propulsion system.

- The overall net thrust target must be delivered by the combination of two new conventional turbofans plus the rear electric fan. The fan is driven by power extracted equally from the LP spools of the two primary engines. The turbofans in your new system must therefore deliver thrust directly aa well as drive the rear fan. There are a large number of design combinations that may be considered for the new primary engines in terms of size or flow rate, bypass ratio, HPT versus LPT trades, turbine stage counts, etc. And the same can be said for the electric fan in terms of flow rate, pressure ratio and diameter. You should examine a select matrix of new hybrid-electric propulsion systems to determine the mass and performance trends in order to select your best candidate.

- The advent of hybrid-electric propulsion has introduced a modern design parameter to define how much power we can extract from a source turbine on either the high- or low-speed spool and also has forced us into re-thinking how we apply our old established design criteria. The *level of power extraction* or *degree of hybridization* (*DoH*) is expressed as a percentage of the source turbine power by

$$\boldsymbol{DoH = \frac{Power\ Extraction}{(Power\ Extraction + Propulsive\ Power)}}$$

where

$$\boldsymbol{Propulsive\ Power = Net\ Thrust \times Flight\ Velocity.}$$

For a conventional turbofan, *DoH* is close to zero, as the power extraction may be considered to be zero.

You need to use this parameter in the design of your new hybrid-electric system. Define DoH as a composed value so you can investigate how it affects performance and net thrust from both primary gas turbine engine and the electric fan in a parametric study.

- Select your best propulsion system based on mass and fuel burn over a simple mission. Define the simple mission based on a typical flight profile for a *Boeing 737-800* or *Airbus A320*. You do not need to calculate installed thrust but please comment on the effect of engine weight on how the overall aircraft weight would modify your calculations and make a statement on the negative effects of installed thrust.

- Run your hybrid-electric system in the off-design mode at static take-off conditions. Document how well the overall net thrust to the aircraft matches that from two baseline engines?

- Please discuss boundary layer ingestion. Why is it attractive? What problems are encountered in its adoption?

It should be noted that the baseline engine model has been constructed at sea-level static take-off conditions where the maximum capable thrust is generated. The value of 24227.5 lbf net thrust in the top right of *Table 3.4* is roughly two times that required by the aircraft for that maneuver and the same may be said for other segments of the mission. This is reflected in the small thrust values in [1], which correspond to actual operations and not to potential maximum delivery of the engines.

	Net Thrust (lbf)	
	Baseline (B)	***Hybrid-Electric (H-E)***
Primary Engine 1	F_B	$F_{H\text{-}E}$
Primary Engine 2	F_B	$F_{H\text{-}E}$
Electric Fan	----	F_{Fan}
Total	$2F_B$ EQUAL	$2F_{H\text{-}E} + F_{Fan}$

Table 4.1: Baseline vs Hybrid-Electric Thrust Requirements

4.2 A Hybrid-Electric System with Rear Fan

Let's think for a few minutes about how we model an electric rear fan, driven by power extracted equally from two primary gas turbine engines. Most preliminary design codes for gas turbine address a single propulsion system any one time, and we are about to handle two turbofans and a fan simultaneously.

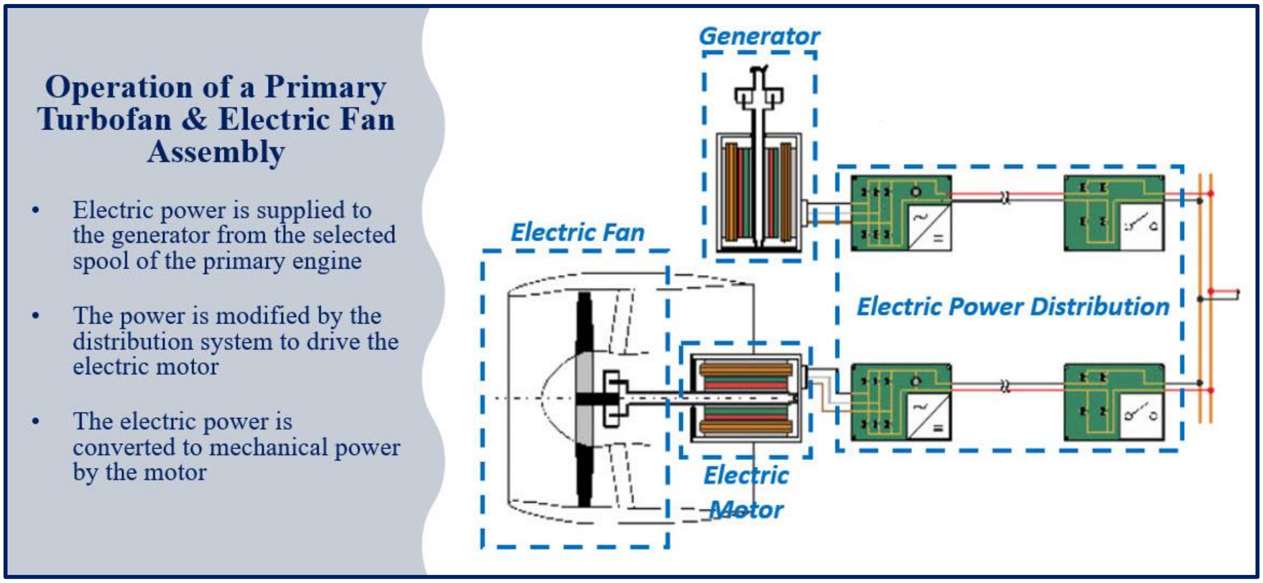

Figure 4.1: Operation of the Electric Fan

Figure 4.1 shows an electric fan assembly using *GasTurb14* and indicates how it works. The generator is connected physically to the LP turbine through the gearbox at the top. The rotational speeds of the LP spool in the primary engine and the generator may be different. We do not pretend to be electrical engineers and we allow our software to handle the electrical components and chose the simple alternative from the two available. You should do this regardless of the software you are using or may have developed yourselves.

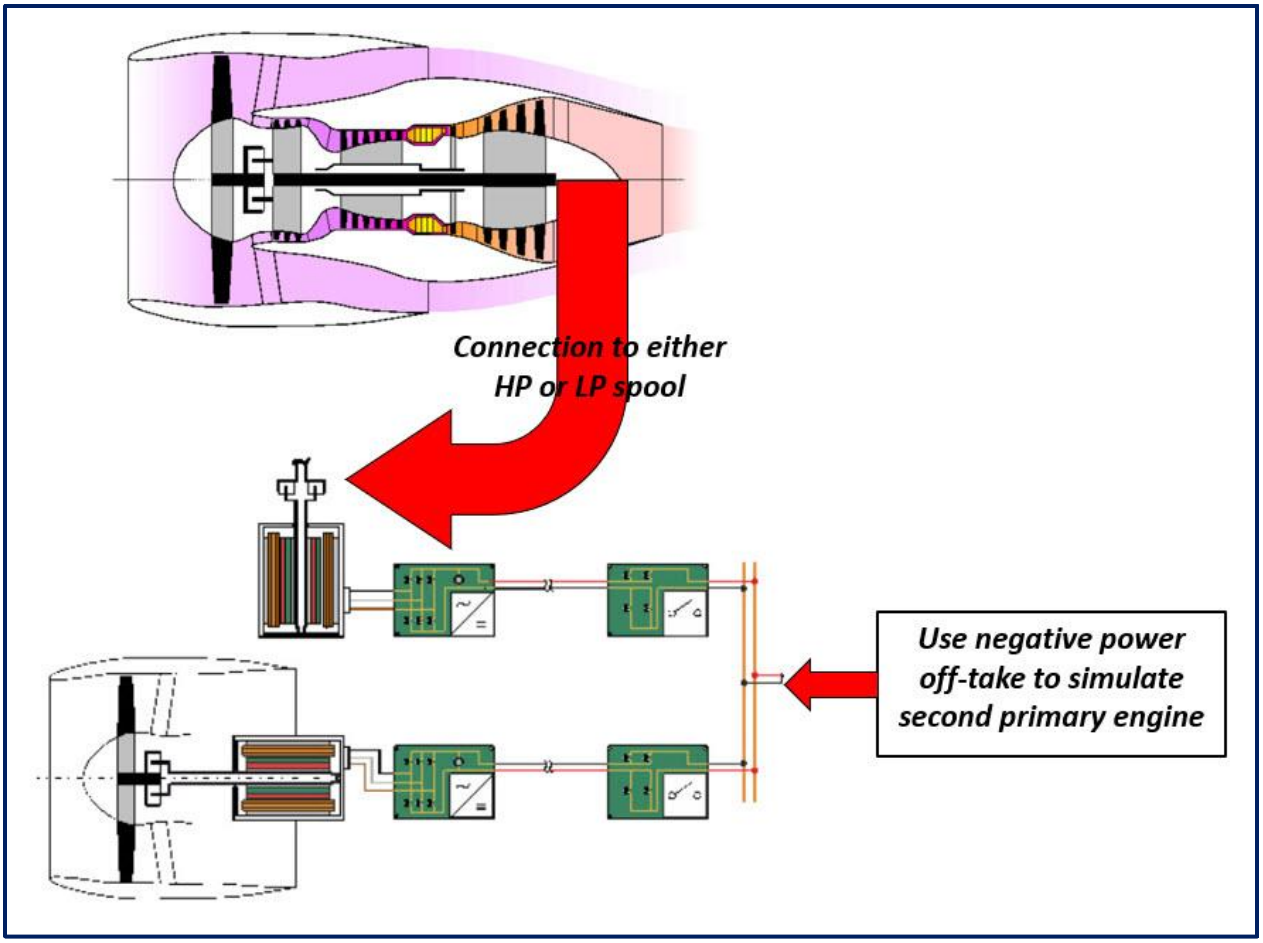

Figure 4.2: Primary Turbofan & Electric Fan Assembly

Figures 4.2 and *4.3* indicate how we accommodate two primary engines in our design scheme. In our hybrid-electric model, we define the LPT power offtake from one of our primary engines as that being fed directly to the generator. To account for the identical input from the second primary turbofan, we set up am equivalent negative power offtake from the electrical system – that is, another input! To ensure that the two distinct types of power input are identical, we set up an iteration within the cycle calculation, which is indicated in *Figure 4.3*.

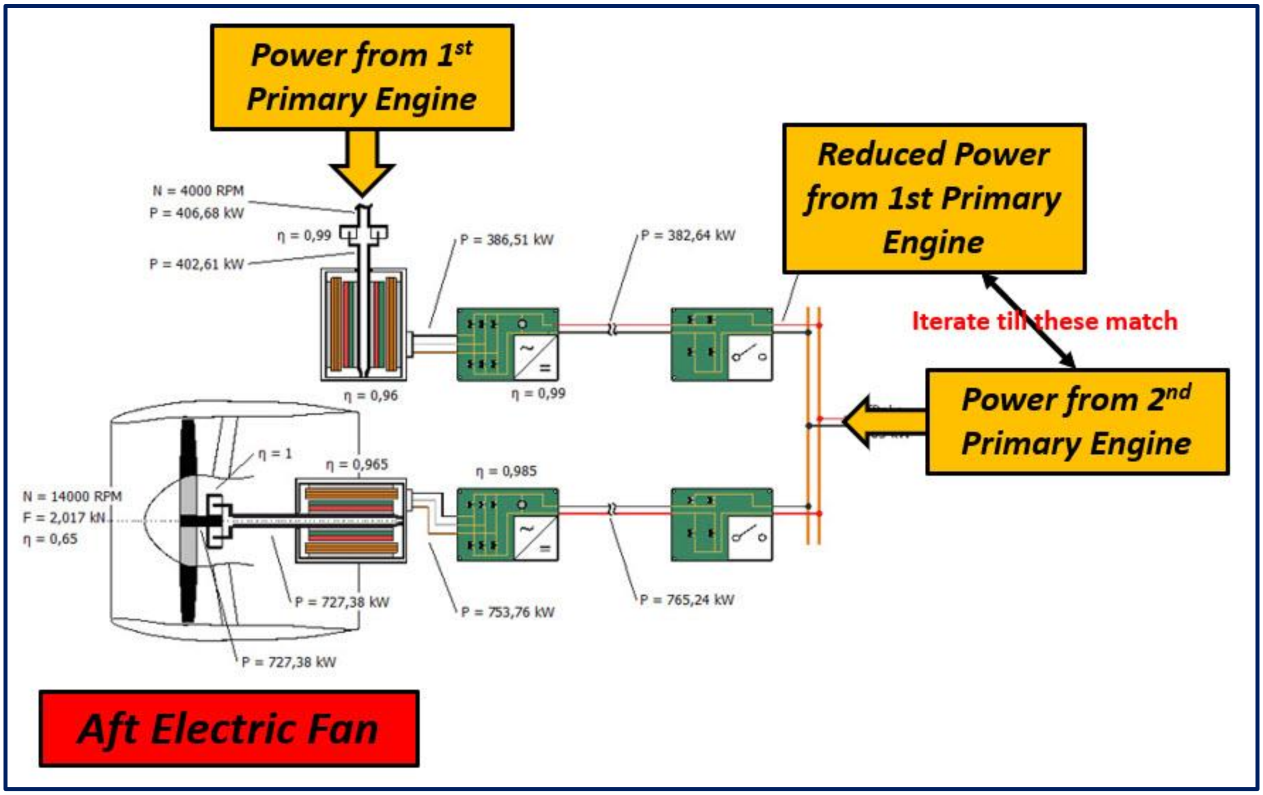

Figure 4.3: Simulation of an Electric Fan Driven by Power Extracted from Two Primary Gas Turbine Engines

4.3 Tutorials

Informal tutorial meetings are offered to all teams who submit the team roster and proposal information form located at www.aiaa-awards.org. A series of meetings can be set up in the fall of 2022 and/or the spring of 2023. These are not pre-arranged and you must contact either of the authors as soon as possible if you are interested. You can ask any questions you like! General advice on running software can also be given.

- You may use any design software which is available to anyone
- The use of design codes from industrial or government contacts, which are not accessible to all competitors, is not allowed.

5. Competition Expectations

The existing rules and guidelines for the *AIAA Foundation Student Design Competition* should be observed and these are provided in the *Appendix*. In addition, the following specific suggestions are offered for the event.

For identification purposes:

NAME YOUR TEAM & NAME YOUR ENGINE

This is a preliminary engine design. It is not expected that student teams produce design solutions of industrial quality, however it is hoped that attention will be paid to the practical difficulties encountered in a real-world design situation and that these will be recognized and acknowledged. If such difficulties can be resolved quantitatively, appropriate credit will be given. If suitable design tools and/or knowledge are not available, then a qualitative description of an approach to address the issues is quite acceptable.

In a preliminary engine design, the following features must be provided:

- Definition and justification of the mission and the critical mission point(s) that drive the candidate propulsion system design(s).

- A clear demonstration that the overall engine performance satisfies the mission requirements.

- Documentation of the trade studies conducted to determine the preferred engine cycle parameters such as fan pressure ratio, bypass ratio, overall pressure ratio, turbine inlet temperature, etc.

- An engine configuration with a plot of the flow path that shows how the major components fit together.

- A clear demonstration of **design feasibility**, with attention having been paid to technology limits.

- Estimates of component performance and overall engine performance to show that the assumptions made in the cycle have been achieved.

While only the preliminary design of major components in the engine flow path is expected to be addressed quantitatively in the proposals, the role of any special secondary systems such be given thoughtful consideration in terms of how it would be integrated into the new engine design. Credit will be given for clear descriptions of how any appropriate upgrades would be incorporated and how they would affect the engine cycle.

Each proposal should contain a brief discussion of any computer codes or *Microsoft Excel* spreadsheets used to perform engine design & analysis, with emphasis on any additional specific features generated by the team.

The page limit for proposals is 50 pages, which will not include the administrative/contents or the "signature" pages.

References

1. *"Conceptual Design of a Single-Aisle Turboelectric Commercial Transport with Fuselage Boundary Layer Ingestion"*
Jason R. Welstead & James L. Felder
AIAA SciTech Forum, San Diego, 2016.

2. "*Aerospace Source Book"*
Aviation Week & Space Technology, January 26, 2009.

3. "*NASA Takes Wraps off Electrical Propulsive Fuselage Concept"*
Aviation Week & Space Technology, October 28 – November 10, 2019.

4. "*Design of Electrified Propulsion Aircraft*"
M. K. Bradley et al
AIAA Short Course, SciTech 2020 – Orlando Jan 2020.

5. *"GasTurb 14: A Design & Off-Design Performance Program for Gas Turbines"*
<http://www.gasturb.de>
GasTurb GmbH. 2021.

6. "*Users' Manual for Updated Computer Code for Axial Flow Compressor Conceptual Design*"
Arthur J. Glassman
NASA Contractor Report 189171, 1992.

7. "*A Simple Correlation of Turbine Efficiency*"
S. F. Smith
Journal of the Royal Aeronautical Society. Volume 69. 1965.

Suggested Reading

1. "Gas Turbine Theory"
   H.I.H Saravanamuttoo, G.F.C Rogers &.H. Cohen,
   Prentice Hall. 5th Edition 2001.

2. "*Aircraft Engine Design*"
   J.D.Mattingly, W.H. Heiser, & D.H. Daley
   AIAA Education Series. 1987.

3. "*Elements of Propulsion – Gas Turbines and Rockets*"
   J.D. Mattingly.
   AIAA Education Series. 2006.

4. "*Jet Propulsion*"
   N. Cumpsty.
   Cambridge University Press. 2000.

5. "*Gas Turbine Performance*"
   P. Walsh & P. Fletcher.
   Blackwell/ASME Press. 2nd Edition, 2004.

6. "*Aircraft Propulsion – Second Edition*"
   Saeed Farokhi
   Wiley, 2014.

7. "*The Jet Engine*"
   Rolls-Royce plc. 2005.

8. “*Propulsion and Power – An Exploration of Gas Turbine Performance Modeling*”
Joachim Kurzke and Ian Halliwell
Springer, 2018.

9. “*A History of the AIAA Undergraduate Engine Design Competition: Its Purpose, How to Write an RFP and How to Win*”
A Tutorial by Ian Halliwell, GTE-01, HSABP-05, INPSI-01. AIAA Propulsion & Energy Forum, August 24, 2020

10. “*Preliminary Design of Gas Turbine Engines”*
A Tutorial by Ian Halliwell. AIAA SciTech Forum 2022 Virtual Presentation.
January 4, 2022

Available Software and Additional Reference Material

"**NPSS® Academic Edition (www.npssconsortium.org):** Numerical Propulsion System Simulation® (NPSS®) proudly sponsors the AIAA Undergraduate Engine Design Competition, with the hope to help students develop valuable skills for the aerospace industry. An academic version of the NPSS software is available for free to all students throughout the world. NPSS is the industry standard for aerospace engine cycle design, analysis, and system integration. Primary applications include aerospace systems, but it can also be used for modeling rocket propulsion cycles, Rankine and Brayton cycles, refrigeration cycles, and electrical systems. A copy of the newly released NPSS Integrated Development Environment (IDE) is available for students participating in the AIAA Undergraduate Engine Design Competition." ***NPSS®***

GasTurb14 is a comprehensive code for the preliminary design of propulsion and industrial gas turbine engines. It encompasses design point and off-design performance, based on extensive libraries of engine architectures and component performance maps, all coupled to impressive graphics. A materials database and plotting capabilities enable a detailed engine performance model to be generated, with stressed disks and component weights. A student license for this code is available directly strictly for academic work. A free 30-day license may also be down-loaded. **(http://www.gasturb.com)**

AxSTREAM EDU™ by SoftInWay Inc. **(http://www.softinway.com)** AxSTREAM® is a turbomachinery design, analysis, and optimization software suite used by many of the world's leading aerospace companies developing new and innovative aero engine technology. AxSTREAM EDU™ enables students to work on the design of propulsion and power generation systems. AxCYCLE™, an add-on to AxSTREAM EDU™ addresses cycle design and analysis. Participants in the AIAA Undergraduate Team Engine Design Competition can acquire an AxSTREAM EDU™ license via the following steps:

- Complete the team roster and proposal information form located at www.aiaa-awards.org

- Once the form has been received and approved, names of team members will be recognized as being eligible to be granted access to the AxSTREAM EDU™ software by AIAA.
- Students must then contact the AIAA Student Competition Chair, listed with the abstract, who will then arrange for SoftInWay to grant the licenses.

In addition to the software, students will also gain free access to STU, SoftInWay's online self-paced video course platform with various resources and video tutorials on both turbomachinery fundamentals.

The offers above are subject to *ITAR* restrictions.

Appendix. Rules and Guidelines

2022 AIAA Foundation Engine Design Competition for Undergraduate Teams

To be eligible for the AIAA Engine Design Competition for Undergraduate Teams, you must complete the team roster and proposal information form located at www.aiaa-awards.org. This information must be submitted by 23:59 hrs. US ET, October 30, 2022 . If you have any questions about the process for submitting this information, please direct them to studentprogram@aiaa.org.

I. General Rules

1. All undergraduate AIAA branch or at-large Student Members are eligible and encouraged to participate.

2. Teams will be groups of **not more than four** AIAA branch or at-large Student Members per entry, unless a larger team has been requested and approved.

3. An electronic copy of the proposal in PDF format must be submitted electronically to AIAA Student Programs. Total size of the file(s) cannot exceed 60 MB, which must also fit on 50 pages when printed. The file title should include the team name and/or university. **A "Signature" page must be included in the report and indicate all participants, including faculty and project advisors, along with their AIAA member numbers**. Designs that are submitted must be the work of the students, but guidance may come from the Faculty/Project Advisor and should be accurately acknowledged. **Graduate student participation in any form is prohibited**.

4. Design projects that are used as part of an organized classroom requirement are eligible and encouraged for competition.

5. More than one design may be submitted from multiple teams of students at any one school.

6. If a design group withdraws their project from the competition, the team chair must notify AIAA Headquarters immediately.

7. Judging will be in two parts.

- First, the written proposals will be assessed by the judging panel comprised of members of AIAA organizing committees from industrial and government communities.
- Second, the best three teams will be invited to present their work to a second judging panel at a special session to be arranged in the *AIAA Aviation Forum*, in June 2023. Scores for the presentations will be combined with those from the written proposals to determine first, second and third places.

8. Commemorative custom-engraved plaques will be presented to the winning design teams for display at their universities and a certificate will also be presented to each team member and their faculty/project advisor. The finishing order will be announced immediately following the three presentations.

II. Copyright

All submissions to the competition shall be the original work of the team members.

Any submission that does not contain a copyright notice shall become the property of AIAA. A team desiring to maintain copyright ownership may so indicate on the signature page but nevertheless, by submitting a proposal, grants an irrevocable license to AIAA to copy, display, publish, and distribute the work and to use it for all of AIAA’s current and future print and electronic uses (e.g. “Copyright © 20__ by _____. Published by the American Institute of Aeronautics and Astronautics, Inc., with permission.).

Any submission purporting to limit or deny AIAA licensure (or copyright) will not be eligible for prizes.

III. Schedule and Sequence of Activities

Significant activities, dates, and addresses for submission of proposal and related materials are as follows:

A. Submit team roster and proposal information form by October 30, 2022
B. Receipt of Proposal – April 1, 2023
C. Proposal evaluations completed - May 1, 2023
D. Round 2 Proposal Presentations & Announcement of Winners at a special session of the AIAA Aviation Forum; date to be decided, in June 2023.

IV. Proposal Requirements

In government or industry, the technical proposal is the most important criterion in the award of a contract. It should be specific and complete. While it is realized that all of the technical factors cannot be included in advance, the following should be included and keyed accordingly:

1. Demonstrate a thorough understanding of the Request for Proposal (RFP) requirements.

2. Describe the proposed technical approaches to comply with each of the requirements specified in the RFP, including phasing of tasks. Legibility, clarity, and completeness of the technical approach are primary factors in evaluation of the proposals.

3. Particular emphasis should be directed at identification of critical, technical problem areas. Descriptions, sketches, drawings, systems analysis, method of attack, and discussions of new techniques should be presented in sufficient detail to permit engineering evaluation of the proposal. Exceptions to proposed technical requirements should be identified and explained.

4. Include tradeoff studies performed to arrive at the final design.

5. Provide a description of automated design tools used to develop the design.

Proposals should be submitted to www.aiaa-awards.org

V. Basis for Judging

Round 1: Proposal

1. Technical Content (80 points)

This concerns the correctness of theory, validity of reasoning used, apparent understanding and grasp of the subject, etc. Are all major factors considered and a reasonably accurate evaluation of these factors presented?

2. Organization and Presentation (10 points)

The description of the design as an instrument of communication is a strong factor on judging. Organization of written design, clarity, and inclusion of pertinent information are major factors.

3. Originality 10 points)

The design proposal should avoid standard textbook information and should show independence of thinking or a fresh approach to the project. Does the method and treatment of the problem show imagination? Does the approach show an adaptation or creation of automated design tools? Focus on an "industrial approach" rather than an academic one.

Round 2: Presentation

Each team will have 30 minutes to present a summary of their proposal to the judging panel with an additional 15 minutes for Q&A. In addition to the categories above, the presentations will be assessed for clarity, effectiveness and the ability to sell the teams' ideas. Scores from the presentation will be added to those from the proposal. The presentation score will be adjusted so that it is worth 30% of the overall value.